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SUMMARY OF EARTH ORBITAL RENDEZVOUS STUDIES

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FOREWORD

Work to date on the Earth Orbital Rendezvous Study has been directed by Department 093-35. The material in this document represents the efforts of E.F. Binz (Department 093-35); D.A. Reed, Jr. (Department 093-38); F.C. Mooney (Department 196-211); and R.E. Freeman (Department 196-231).

This document is in two volumes. The first, SID 62-834-1, is unclassified; the second, SID 62-834-2, is classified Confidential. Some references in Volume 1 are to figures contained in Volume 2.

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I. INTRODUCTION

The objectives of Project Apollo are to place three men in earth orbit, in circumlunar orbit, and on the moon and to safely return them to earth in the 1960 to 1970 time period. Accomplishment of these objectives will involve the solution of two major problems, one relating to boost system size versus payload requirements and the other relating to the versatile utilization of existing equipment and existing concepts.

The Saturn boost systems from C-1 through C-5 have been found wanting in payload capability for the objective of placing three men on the moon. It is recognized that the Nova-class booster systems will be most effective for a continuous-trajectory, single-propulsion-system approach to this objective. Political pressures, problems of cost, and the boost system development time span will probably involve the use of other means than Nova to accomplish all of the objectives of Project Apollo within the 1960 to 1970 time span. In many cases, cost problems will require the use of existing equipment.

Alternatives to the Nova boost systems include

1. Earth-orbital rendezvous of two C-5 payloads
2. Lunar-orbital rendezvous of a single C-5 payload that includes a three-man Apollo vehicle and a two-man lunar excursion vehicle.
3. Earth-orbital fueling or refueling

It is the purpose of this study to investigate the merits and capabilities of the earth-orbital rendezvous (EOR) of two C-5 boost system payloads to accomplish the Apollo objective of placing three men on the moon and safely returning them to earth.

POSSIBLE MODES OF EOR

The possible modes of EOR are the following.

Orbital Transfer

Assembly Mode

The first C-5 launches an S-IVB into target orbit ($<$ 300 nautical miles). The second C-5 launches the spacecraft into parking orbit ($>$ 100 nautical miles). The spacecraft transfers orbits, docks, and mates.

The first C-5 launches an S-IVB into parking orbit. The second C-5 launches the spacecraft into target orbit. The S-IVB transfers orbits, docks, and mates.

Tanking Mode

The first C-5 launches an LO_2 tanker into target orbit. The second C-5 launches the assembled S-IVB (less LO_2) and the spacecraft into parking orbit. The assembled vehicle rendezvous with the tanker and fuels the S-IVB.

The first C-5 launches an LO_2 tanker into parking orbit. The second C-5 launches the assembled S-IVB (less LO_2) and the spacecraft into target orbit. The tanker rendezvous with the assembled vehicle and fuels the S-IVB.

Direct Ascent

The first C-5 launches an S-IVB into an orbit of approximately 125 nautical miles. The second C-5 launches the spacecraft to rendezvous with the S-IVB in its orbit.

EOR WORK OUTSIDE S&ID

A comprehensive bibliography on rendezvous is presented in SID 62-559, Space Rendezvous: Technical Documentation.

Vought-Astronautics has made a study of orbit launch operations under Contract No. NAS 8-853, for MSFC, with support from Norair, Raytheon, AMF, Douglas, IBM, and Sperry-Rand.

MSFC has published MTP-CP-62-1, Volumes 1-5, Earth-Orbital Operations, June 1962.

II. EARTH-ORBITAL RENDEZVOUS ORBITAL MECHANICS

This study was made in an effort to determine the capability of the C-5 combinations of S-I, S-II and S-IVB to achieve an earth-orbital rendezvous and the capability of the Apollo configuration to accomplish a lunar mission. The feasibility of such a launch configuration is dependent upon the velocity impulse or propellant weight required for orbital transfer, rendezvous, and docking maneuvers. Considerable effort has been expended to define these requirements by parametric analyses of launch timing, orbit choice, and vehicle phasing. The major results of this study are the following:

1. The lunar landing module should be enlarged to the maximum C-5 orbit capability to allow a maximum maneuvering propellant weight.

2. Launch delays of at least two hours for both vehicles are tolerable; adjustments in launch azimuth only are required.

3. A maximum of 12 waiting orbits could be required for the worst launch delay time.

4. The rendezvous maneuver could be accomplished with a 1000 to 1200 ft per sec velocity impulse, which is within the C-5 payload capability.

For the study, it was assumed that it would be possible for the two payloads to arrive in the lunar launch window at a prescribed time after rendezvous; therefore, this consideration was eliminated from the study since it was only a matter of mission planning.

BOOSTER PERFORMANCE

An analysis of the performance capabilities of the first two stages of the C-5 system indicates that the maximum payload capability varies as a function of end boost altitude. For a direct ascent to high altitude orbits, a payload loss is incurred. This performance capability is illustrated in Figure 2-1.

If we translate the data in Figure 2-2 into ideal velocity impulse available, we are able to see how much excess velocity we have to complete the rendezvous, docking maneuvers, and the subsequent injection into lunar orbit.

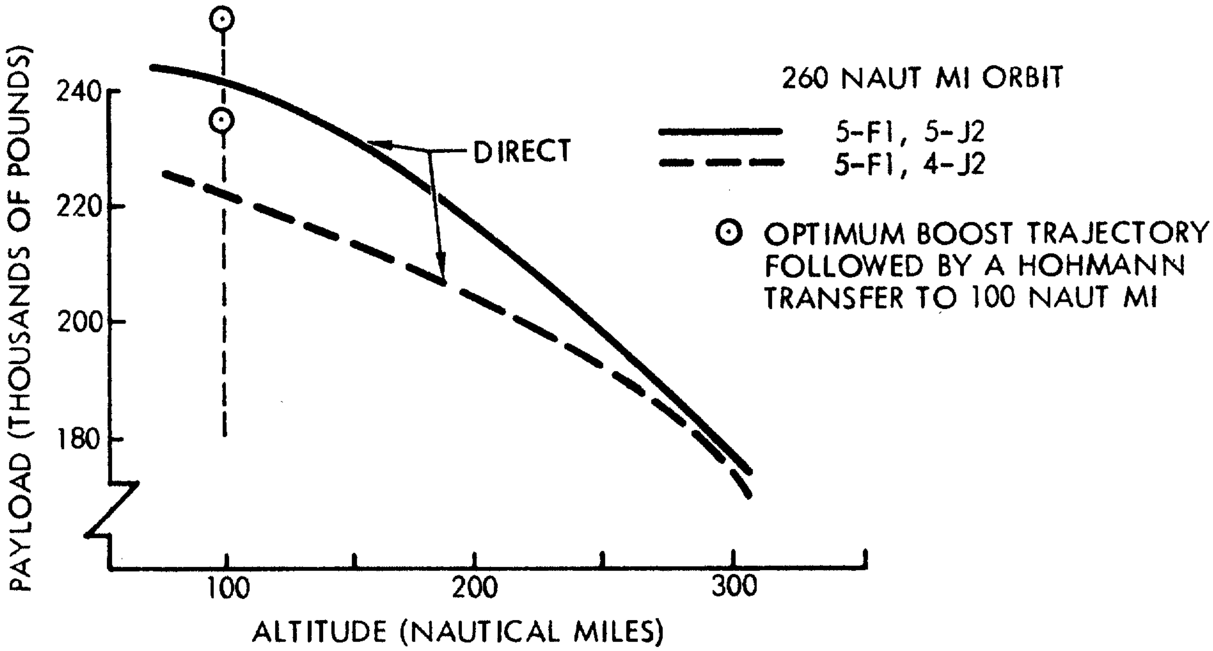


Figure 2-1. Payload Capability Versus End-Boost Altitude

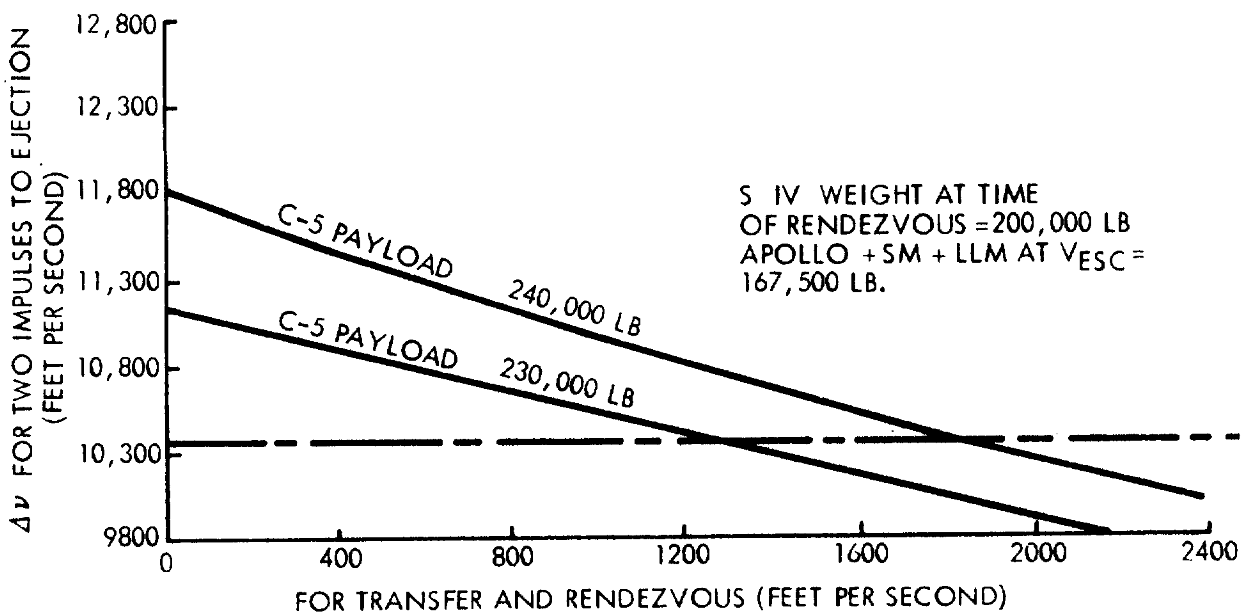


Figure 2-2. Rendezvous and Lunar Ejection Performance Capability (Two Stages to Ejection)

These data are given for C-5 low earth-orbit payload weights of 230,000 and 240,000 lb. The S-IVB weight at time of rendezvous was assumed to be only 200,000 lb. Since the S-IVB structure information and possible extra insulation requirements were not available to S&ID the rate of fuel loss due to boiloff could not be determined. The 40,000-lb weight loss, which includes orbit transfer requirements and boiloff, is assumed reasonable for a stay in orbit from three to seven days; this stay should be sufficient for any foreseeable Apollo launch delay.

GROSS RENDEZVOUS PROPULSION REQUIREMENTS

The study of the propulsion or velocity requirements of orbit transfer was based on certain simplifying assumptions. An ideal velocity impulse was assumed at two points, one lying on an initial 100-nautical-mile circular orbit and the other lying on a final 300-nautical-mile circular orbit. Although two impulse orbit transfers do not provide the entire answer, they are much simpler to analyze and are probably more feasible from an operational standpoint. The results obtained from two impulse studies are, in most respects, conservative. It was realized that the 300-nautical-mile final circular orbit was higher than that recommended for manned operations; but since the study was conceptual, this altitude was chosen to allow for the maximum separation between initial and final orbit altitudes. In a detailed study on orbit transfer, there would certainly be a tradeoff between the orbit altitude difference, phase relationships, and transfer windows.

The 100-nautical-mile initial circular orbit was chosen for minimum orbital decay and high booster payload capabilities.

The launch timing portion of the study indicated that some angular (inclination) difference between the two orbital planes was probable, and this difference was considered first. The minimum impulse required to make a transfer from 100 to 300 nautical miles with an out-of-plane angle is presented in Figure 2-3.

These transfers generally begin and end near the intersection (i.e., the node) of the two orbital planes. The chart shows that for reasonable velocity impulse, the out-of-plane angle should be kept as low as possible.

In light of this requirement, the departure and arrival geometry of the transfer maneuver was investigated to determine the phase angles required for rendezvous. Figure 2-4 presents a map of the velocity impulse surface as a function of interceptor position from the node and target angle at the start of transfer for a given out-of-plane angle of 1.5 degrees. The inset illustrates the coordinates of this figure.

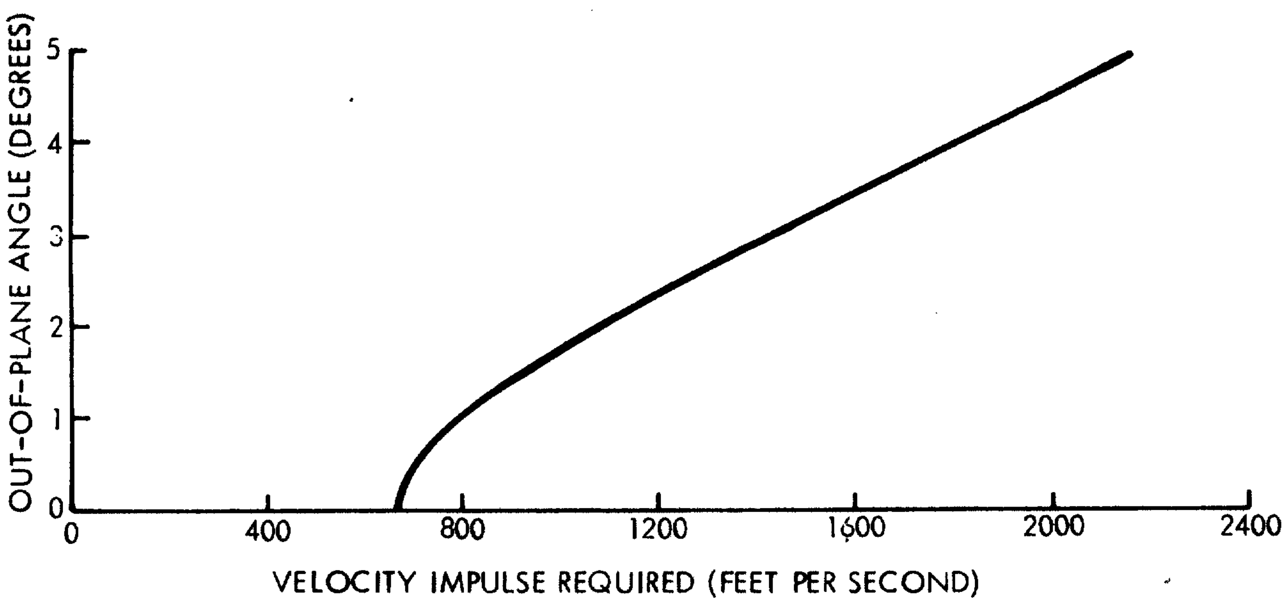


Figure 2-3. Minimum Impulse Versus Out-of-Plane Angle (100 to 300 Nautical Mile Transfer)

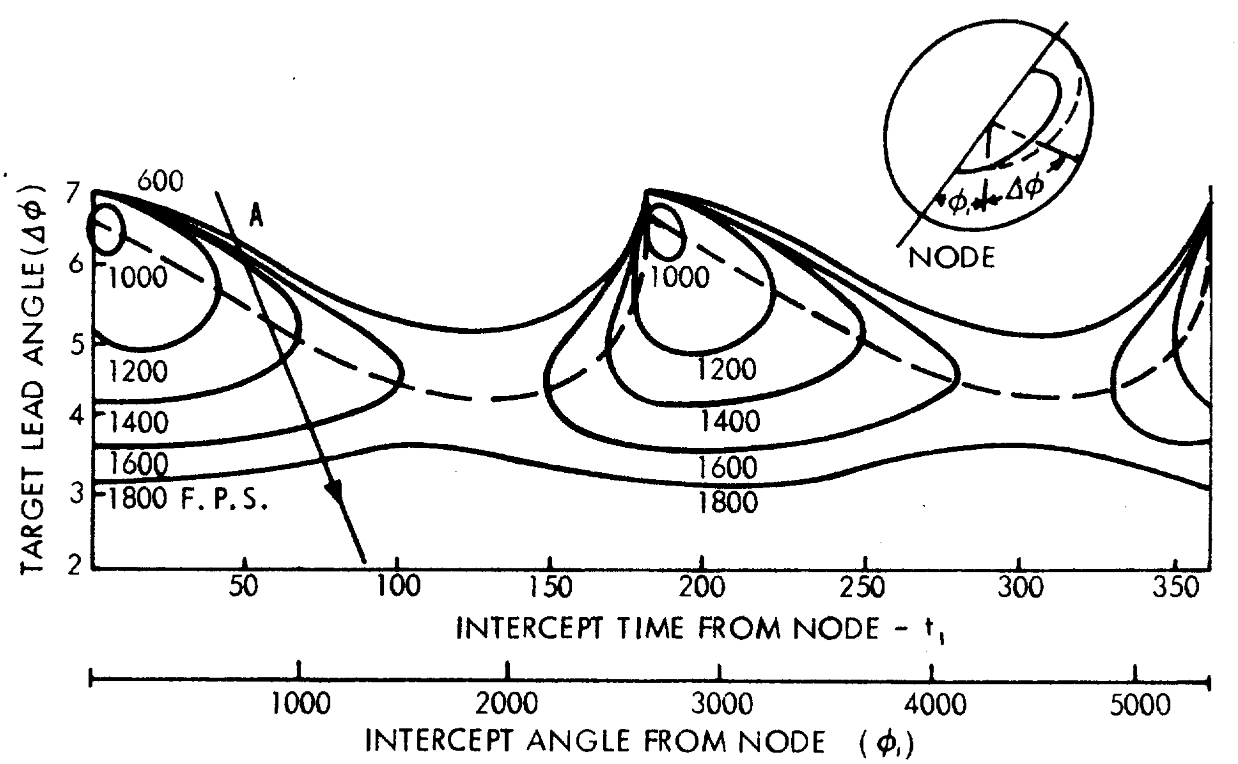


Figure 2-4. Velocity Impulse Surface (Out-of-Plane Angle = 1.5 Degrees)

An example of a typical track of the two vehicles through this geometry is shown as the vector A (Figure 2-4). The slope of this line is directly related to the difference in altitude of the two orbits; and it determines the time window size for making the transfer within a given impulse, since the ordinate in the figure is directly related to time. The dashed curve gives the configuration required at departure for the minimum impulse transfer. Additional data are presented in Figure 2-5 for out-of-plane angles of 3.0 degrees.

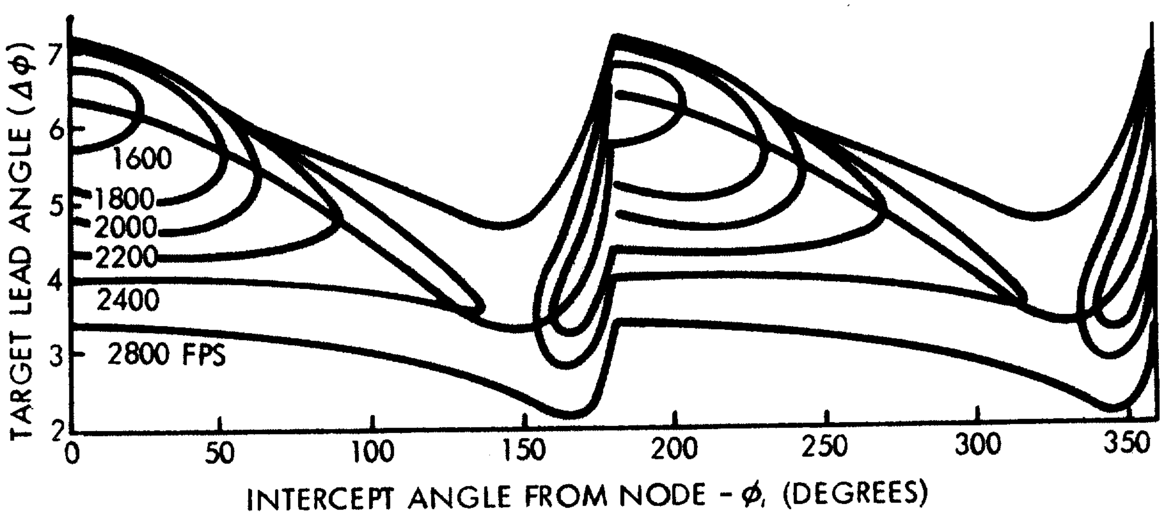


Figure 2-5. Velocity Impulse Surface (Out of Plane Angle = 3 Degrees)

An illustration of the range angles associated with the transfers shown in Figure 2-5 is presented in Figure 2-6 as velocity impulse versus transfer range angle for various values of departure angle.

These data illustrate the penalty paid for any deviations from a quasi-Hohmann (i.e., 180 degrees), node-to-node transfer. This penalty is largely dependent on out-of-plane angles; therefore, for lower out-of-plane angles, the range angle might possibly be adjusted to provide flexibility in phasing and to remain within a given velocity impulse range.

LAUNCH TIMING

One approach for launching two vehicles is the simultaneous launch of both payloads into the same orbit. Many factors favor this approach, but it is beyond the capabilities of the C-5 launch complex at AMR, where multiple

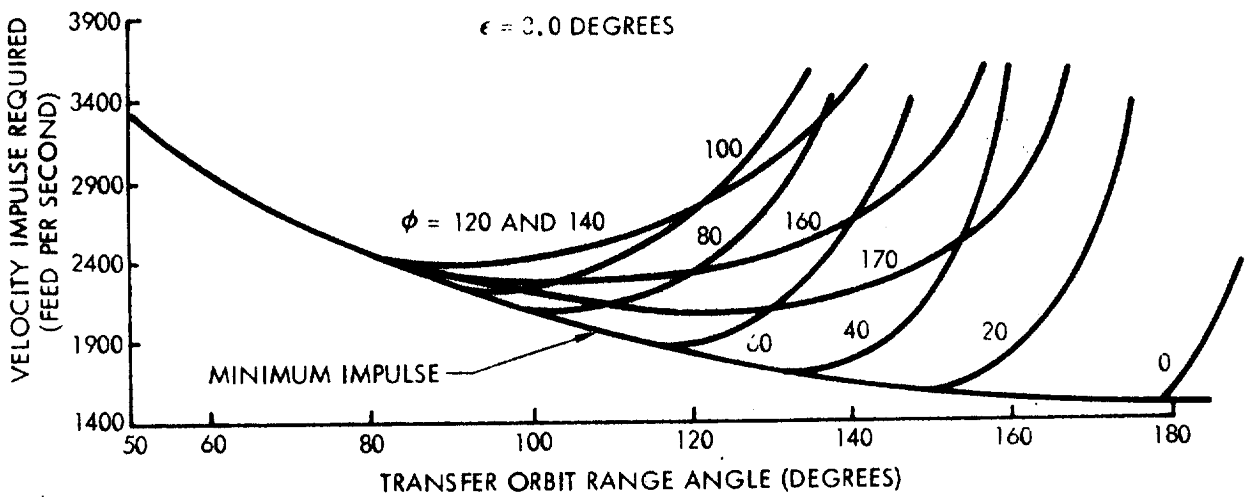


Figure 2-6. Velocity Impulse Versus Transfer Range Angle for Various Values of Departure Angle

launchings are restricted to a time delay of at least 1 to 3 hours; a catastrophic failure in one vehicle would probably result in the loss of both. The next approach considered is a launch of the first payload into an orbit that will cross the launch latitude the same day at a later time. An example of this type of orbit is given in Figure 2-7

The coordinates of this figure are space coordinates; therefore, the orbit track remains fixed, and the launch point moves on the launch latitude as some function of time, which is approximately equal to the earth's rota-rotational rate (15 degrees per hour or 360 degrees per day). The angle between launch point 1 and launch point 2 is a function of the orbital inclination and is directly related to the time required for the launch point to move from point 1 to point 2.

If it should be desirable to launch the second payload the same day at some later time (T + dt), the first payload orbit could be placed so that it would cross the launch latitude the same day at point 2. The angular difference between point 1 and point 2 would relate directly to the time delay desired.

The equations required to calculate the orbit inclination required to achieve the geometry of Figure 2-7 are the following:

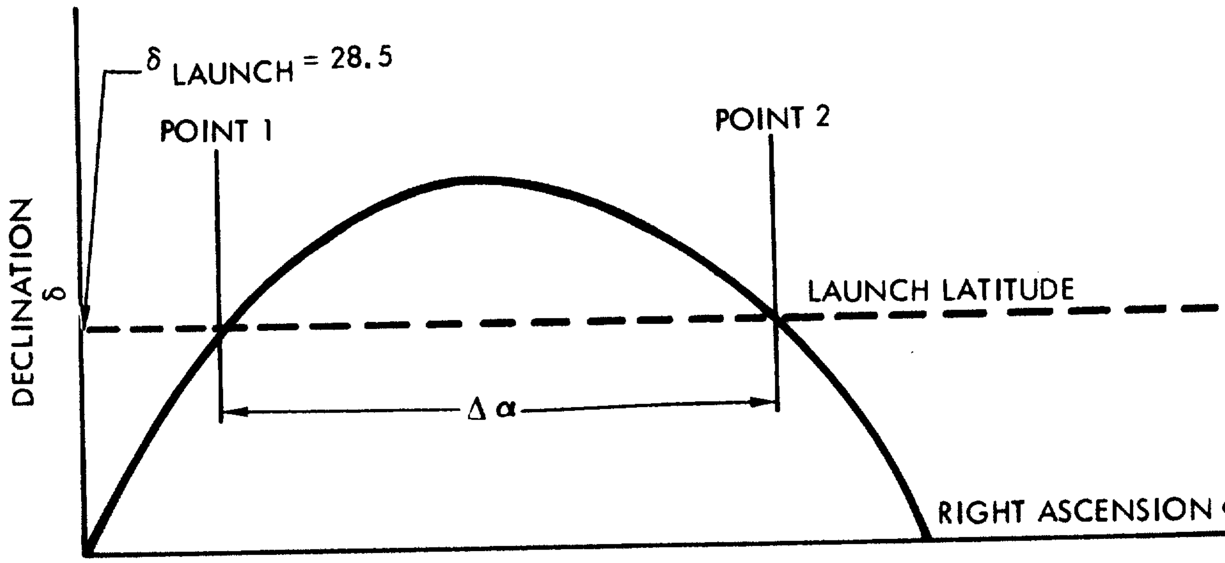


Figure 2-7. Typical Orbit Track

$$d\alpha = \left[90 - \frac{dt \times 15}{2} \right]$$

$$\text{Inclination} = i = \text{ARC TAN} \left[\frac{\text{TAN } (\delta \text{ LAUNCH})}{\text{SIN } (d\alpha)} \right]$$

These equations hold for less than 12 hours, and it can be seen that the orbit inclination required for large time increments between launches approaches 90 degrees or polar orbit.

Another concept of launch timing is the general case, that is, the second payload launched at some time greater than 12 hours following the launch of the first payload. Many of the topics discussed here will also apply to the concept of the second launch the same day, at some later time.

Figure 2-7 presents the first vehicle orbit in space coordinates. Thus far, this section has been concerned with the effect of launching the second vehicle, or interceptor, at a given time following launch of the tanker. In the general case, this time separation could be a number of days.

In the coordinates of Figure 2-7, the launch point moves on the launch latitude. If the launch point is assumed to be X, in the Figure, launch could be made into an orbit that would intersect the target orbit.

The angle of intersection would be quite large and, as stated in the discussion of orbital transfer, would require large amounts of velocity impulse. The most desirable launch point of the tanker would be either at point 1 or at point 2 in a norteasterly or southeasterly direction. These launch times would result in the two orbits being co-planar with the least velocity impulse required for the orbit transfer. A second concept of launch delay times involves delays from a present time for launch caused by weather conditions in the launch and abort recovery areas and equipment malfunctions and, not by the C-5 complex operation. These delays may be a matter of hours. In this discussion, it is assumed that the preset launch time is chosen when the launch site is at point 1, so that an in-plane launch is possible. A delay of launch from this time will cause the interceptor orbit to lie out of the plane of the tanker orbit by some angle ϵ, unless that delay causes the launch to occur at point 2 and makes a coplanar orbit again possible. A chart of the effect of the delay times on minimum out-of-plane angles is presented in Figure 2-8.

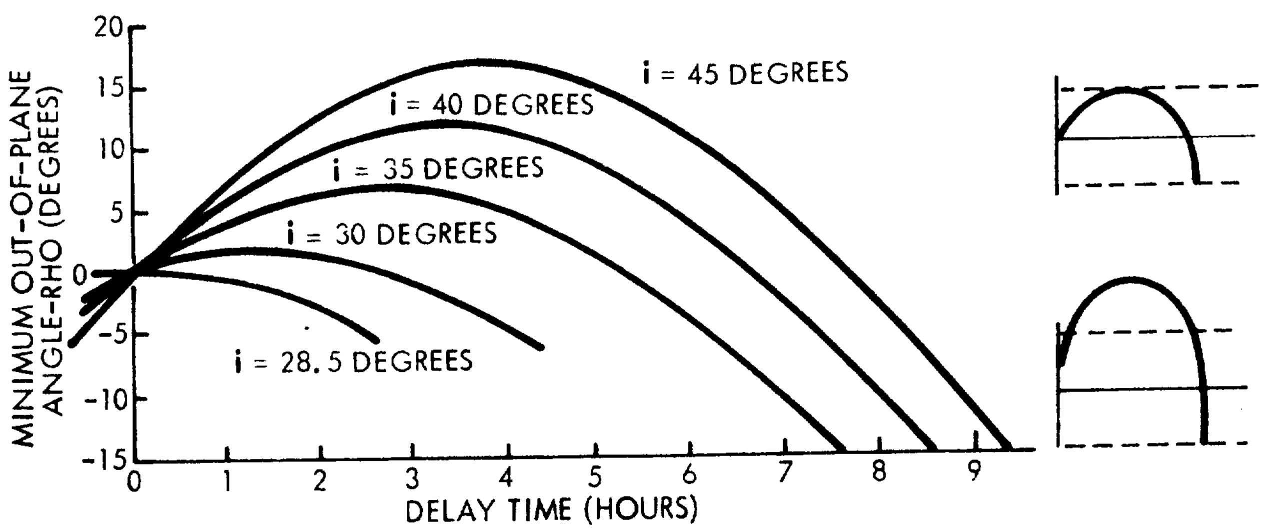


Figure 2-8. Effect of Delay Times on Minimum Out-of-Plane Angles

Figure 2-8 shows that two types of launch windows exist for a given maximum value of ρ. These are continuous and discontinuous, and they are illustrated in the insets A and B. Without exceeding a given value of ρ, the continuous window characteristics allow for a launch at any point on the time scale. The discontinuous window is actually two windows, one about the northeasterly in-plane launch and the other about the southeasterly in-plane launch.

This document concentrates on the continuous launch windows since the inclination desired for the Apollo lunar orbit is the lowest available from AMR and since those inclinations that produce reasonable values of minimum out-of-plane angle also produce sufficient delay times. The inclination that results in a 2-hour window is 29.4 degrees.

After target vehicle orbit has been established in a given inclination based on the desired minimum out-of-plane angle and the required launch delay capability, launch azimuth requirements and phasing for rendezvous will be considered, i.e., the location of the target vehicle in its orbit at the time the interceptor is launched. To achieve the minimum out-of-plane angles previously discussed, the interceptor orbit must intersect the target orbit at a range angle (β) of 90 degrees, which corresponds to a given firing azimuth angle. The effect of the target position on the intersection point, or node, at the time of launch is shown in Figure 2-9, which presents

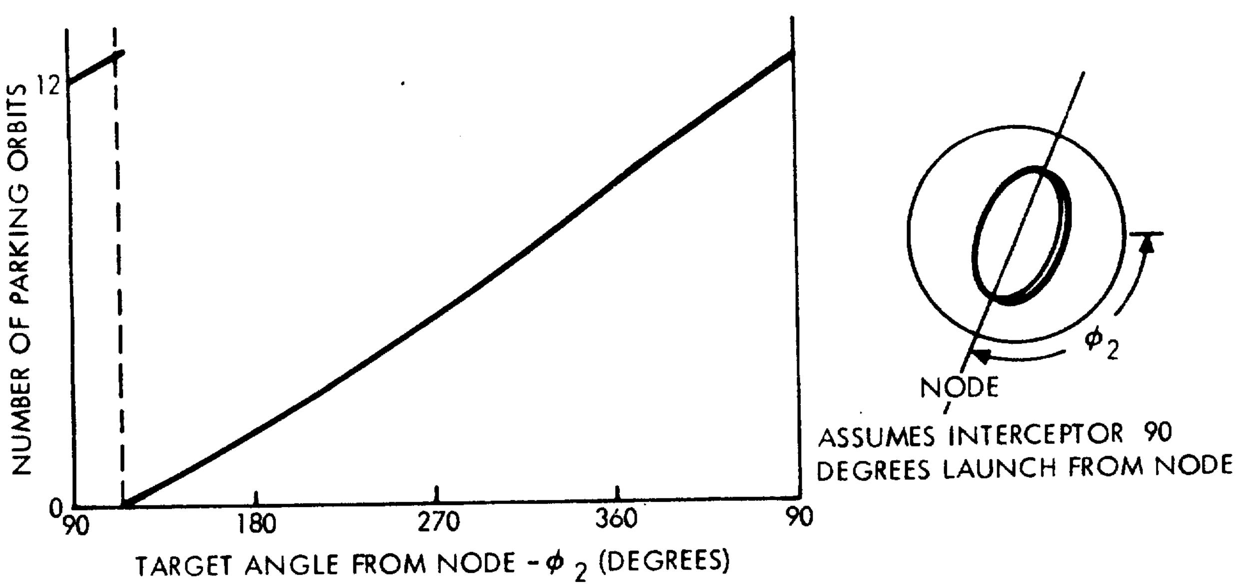


Figure 2-9. Waiting Time Characteristics

the number of parking or waiting orbits required to reach a configuration that will allow rendezvous with a reasonable velocity impulse.

In Figure 2-9, it can be seen that a maximum of 12 waiting orbits could be required before the transfer and rendezvous would be possible. It is also shown that if the vehicle is approximately 94 to 97 degrees from the node, the transfer may be made immediately. It is this case that we refer to as the direct rendezvous.

An analysis of the direct rendezvous is warranted since it is an attempt to shorten the waiting time in orbit. If the target position is such that a range angle of other than 90 degrees is indicated for a direct rendezvous, this mission will result in an out-of-plane angle of greater value and probably in a greater impulse to achieve the rendezvous transfer.

The effect of range angle on out-of-plane angle is presented in Figure 2-10 as a function of the minimum ϵ or ρ.

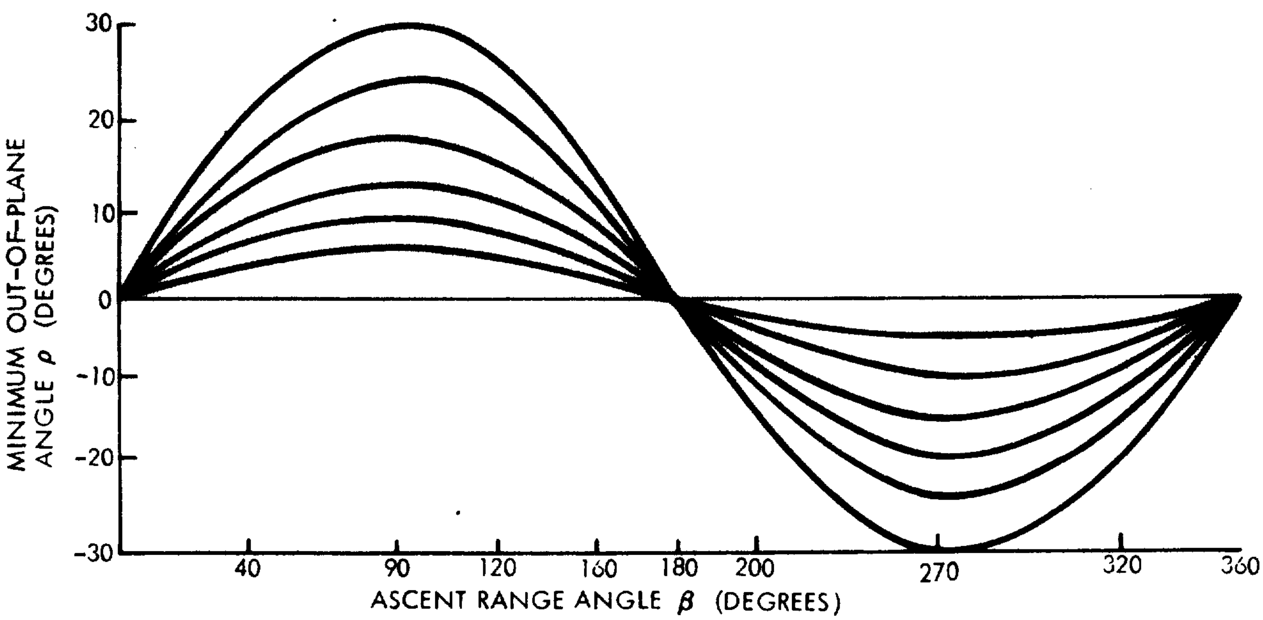


Figure 2-10. Variation of Minimum Out-of-Plane Angle with Ascent Range Angle

These data show that for a given value of ρ or minimum ϵ, the range angle deviation from 90 degrees will determine the actual out-of-plane angle. For the small values of ρ being considering herein, a range angle variation of 45 to 135 degrees results in out-of-plane angles up to about 2 or 3 degrees, which are reasonable. With reference to the effect on transfer velocity impulse of this configuration, an example of which is presented in Figure 2-6, it can be seen that the increase over minimum is approximately 2000 ft per sec required for the 45 or 135 degree range angle intercepts is 3600 ft per sec.

Those direct intercepts that are not far from minimum impulse (i.e., β = 90 degrees) do not result in large velocity requirements, and it is possible to use this type of transfer to change the waiting time in orbit before rendezvous. This would be accomplished by allowing a small increase in out-of-plane angle, while changing the initial phase angle of the target and interceptor vehicles. Since the allowable change would be only 10 or 20 degrees, the total number of waiting orbits could not change; but the point in the orbit where the rendezvous configuration is achieved could be moved to allow a lower impulse transfer or a larger transfer window for a given velocity impulse.

III. PRELIMINARY DESIGN

ASSUMPTIONS AND CONCEPTS

Approximately 400,000 to 500,000 pounds of payload in a 480-kilometer earth orbit are required to support the three-man Apollo vehicle to travel to the moon, land on the moon, and return to earth. The precise value of payload is dependent, in part, upon over-all system efficiency and the propulsive capability of available fuels. Further, the effect of the extreme reliability on man-rated components and of the margins for error incorporated in the vehicle design widen the payload requirement range. Since efficient, man-rated boost systems of raw power do not exist, reliability and margins for error play a predominant role in this investigation.

Some of the broad assumptions guiding the design effort of earth orbital rendezvous are the following:

1. Maximum component reliability
2. Incorporation of the maximum possible margins for error allowances for propulsion and component design
3. Multiple utilization of components and equipment
4. Restriction of propellants to the highest energy content consistant with multiple sybsystem utilization and state-of-the-art reliability. (State of the art shall mean concepts capable of reliable application during the 1960 to 1970 time reference. This does not restrict the utilization of exceptional ideas where evaluation deems full possibility of utilization.)

Some lesser assumptions include the utilization of the C-5 payloads as third stage boosters. Additional payloads in orbit can be realized in this way. Further, the utilization of a two-stage translunar injection is included so that more payload can be placed in the vicinity of the moon. The multiple staging of lunar landing modules has also been considered.

Two basic rendezvous concepts have been explored. The first involves two C-5 payloads. One payload consists of (1) an Apollo command module, (2) an Apollo service module, (3) a lunar landing module, and (4) a translunar injection booster fueled with LH_2 and containing an empty tank for

LO_2. The second C-5 payload is an S-IVB booster fueled with an excess of LO_2. The excess LO_2 is to be transferred to the first empty C-5-payload LO_2 tank during an earth-orbital rendezvous operation.

The second basic rendezvous concept also involves the utilization of two C-5 boost system payloads. The first C-5 payload is a fueled S-IVB placed in a 480-kilometer parking orbit and is equipped with a Saturn instrument unit module and a docking and thrust load capability in its nose cone. The second C-5 payload consists of an Apollo command module, an Apollo service module, a lunar landing module and a docking and translunar booster thrust structure.

Other means for rendezvous, such as impact bag docking structure, parallel approach trapeze systems, off-axis tanker probe systems, tandem tanker systems, and S-II utilization as an LO_2 tanker, were considered. Each method studied for rendezvous had certain deficiencies either in performance, payload capability, or damage susceptibility. Combining the best features of these rendezvous concepts has resulted in a concept in which a translunar booster can be mated to Apollo or tanker fuel transfer can be provided in space. Booster/payload trajectory optimization studies become, therefore, the most significant factor in the accomplishment of Apollo objectives. Rendezvous requirements are severe; however, the simplicity of the rendezvous concept permits the utilization of two- or three-stage boosts into earth orbit, one- or two-stage translunar injection boost programs, and multistage lunar-landing functions without significant changes in rendezvous hardware application.

ASSOCIATED PROBLEM AREAS

Some of the problems related to space rendezvous are the following:

1. Acquisition and tracking in space of objects to be mated
2. Diverse power/thrust spectrum during boost, orbit transfer, attitude control, and maneuvering
3. Dynamic behavior of large masses impacting in zero g fields
4. Damage sensitivity of systems mated
5. Orbit station keeping and maneuver dynamics
6. Retention of cryogenics with low loss over protracted time periods in the environment of space

7. Behavior of materials in space environments

8. Fundamental system reliability and the associated equipment reliability

9. Behavior of fluids in zero-g fields

10. Solar radiation effects on men and material

11. Particle penetration and means of protection against penetration damage

12. The man/machine system behavior

13. Hardware development costs and man-rated producibility

14. Payload configuration, size, and utilization

It is obvious that mission objectives and the payload configuration will be the deciding factors in the solution of a given rendezvous operation related to mechanical components utilized for in-space mating of large masses. Considerations of system efficiency will be dominated by the reliability of the man/machine complex and the associated protection against the harsh environment of space, since nonproductive payload allowances must be made. The solution of the related problems associated with man-in-space is not a part of this study. However, as far as practical, these problems are considered in light of their influence on vehicle size, configuration, and docking capabilities. The earth-orbital rendezvous procedures described herein will include the capabilities of the Saturn C-5 boost system, the Apollo mission (man on the moon with safe earth return), the effect of space environment, reliability, and versatility of equipment proposed.

MISSIONS AND TRAJECTORIES

The descriptions of trajectory dynamics, mechanics, mission objectives, and related analytical solutions will be described in reports of associate group effort. The effects of these trajectories and missions upon the rendezvous procedure, equipment utilization, and system requirements are included here to establish reasonable areas of investigation for design of rendezvous hardware.

The Saturn C-5 two- and three-stage boost-to-earth-orbit systems are assumed for earth-orbital rendezvous payloads. However, for the three-stage boost systems, the payload is further assumed to supply the third stage boost function. The reason for assuming a boost function for the payload is that additional constraint placed on the design of rendezvous hardware may be too severe.

The Apollo lunar landing module has been assumed to function as (1) a third-stage booster to an earth orbit of 225 kilometers, (2) a low earth orbit to high earth orbit transfer booster, (3) a second-stage translunar injector booster, (4) the translunar midcourse velocity correction booster, (5) a lunar retro booster to lunar orbit, and (6) a lunar retro and landing module. In conjunction with the Apollo service module, all Apollo attitude control functions and earth-orbital rendezvous and docking maneuver power is derived from the service module and the lunar landing module. So many functions incorporated in a single module may not yield the best possible performance, but the assumption is logical in view of the fact that the Apollo must be man-rated: this, in turn, implies utilization of previously qualified systems, simplicity of component design, and flexibility of system and component operation.

The S-IVB booster payload has much simpler functions to perform. The booster has been assumed to operate as (1) a third-stage booster to place itself in a 480-kilometer earth orbit; (2) a first-stage translunar injection booster; or (3), in some operations, an in-space fuel tanker.

A typical manned lunar-landing and earth-return mission might be the following:

1. An S-IVB boost stage is placed in a 480-kilometer earth-parking orbit with maximum possible propellant loading at the highest possible mass fraction. The Saturn C-5 is assumed to be the boost system.

2. The S-IVB booster is then oriented to be relative to solar rays to conserve propellants.

3. An Apollo, previously described, is placed into low earth orbit of 225 kilometers.

4. After acquisition and adequate tracking of the S-IVB booster by the Apollo, the Apollo lunar landing module vernier engines are fired to provide the necessary velocity increment for orbit transfer.

5. The orbit transfer of the Apollo is timed to permit arrival in the 480 kilometer orbit above and ahead of the S-IVB. The Apollo orbit injection and coplanar thrust is applied on a radar range

decrement and relative velocity differential (range-to-go) basis relative to the S-IVB.

6. Station keeping at 500 to 1000 feet is established by Apollo fly-by-wire commands of the astronaut pilot.

7. The astronaut activates the docking system on the S-IVB by radio command (See Drawing 7121-1901-2) and the homing transmitter in the docking cone structure attached to the lunar landing module landing gear.

8. The astronaut maneuvers the Apollo into axial alignment with and off the nose of the S-IVB. Vidicons located on the Apollo docking structure confirm orientations (as do telemetog signals of the S-IVB boom system).

9. The astronaut commands thrust to maneuver the Apollo toward the nose of the S-IVB. During the approach, the S-IVB docking automatically homes on the central zone of the docking cone located on lunar landing module landing gear.

10. When the docking boom impacts and locks to the cone, the excess relàtive-motion energy is absorbed by the boom in axial motion, and the pitching motions of the Apollo and the S-IVB are restrained by their respective automatic attitude control systems.

11. The docking boom is commanded by radio to retract and lock. The relative approach of Apollo and S-IVB places the longitudinal axis of the S-IVB into co-axial alignment with the longitudinal axis of the Apollo in random roll orientations. The Apollo docking cone and the S-IVB nose cone nest and provide a thrust and bending socket structural connection.

12. The instrument unit guidance platform on the S-IVB is slaved to the Apollo guidance platform via radio link. In general, it is preferred that radio link operation is utilized so that mating complexity is minimized.

    The physical alignment of the Apollo and the S-IVB longitudinal axes and the coincidence alignment of the Apollo and instrument-unit, guidance-platform guidance vectors permit precise control of the S-IVB thrust vector and attitude control forces and moments by the Apollo astronaut or automatic control equipment. The S-IVB platform operation is that of a vector resolver unit when it is mated and slaved to the Apollo platform.

13. The combined Apollo/S-IVB is maneuvered to translunar injection attitude and placed on automatic boost sequence or optional astronaut pilot-command sequence.

14. At the proper time and orbit location, the S-IVB fires and functions as a first-stage translunar booster until propellant burnout.

15. The docking structural release locks located on the Apollo lunar landing module landing gear are released, and the landing gear is extended to lunar landing attitude. (See Drawing 7121-1901-2.)

16. The lunar landing module is now fired to translunar cutoff velocity.

17. At translunar midcourse, the Apollo is properly oriented, when necessary, and the lunar landing module vernier engines are fired for midcourse correction.

18. On approach to the moon, the Apollo vehicle is properly oriented; and the lunar landing module main and vernier engines are fired to establish a low lunar orbit.

19. After a reasonable time in lunar orbit, the lunar landing requirements are established; and the lunar landing module engines retro fire and land the Apollo and the lunar landing module on the moon.

20. Lunar takeoff and earth return is accomplished with the service module propulsion system.

It is appropriate to underscore here that the Apollo/lunar-landing-module and S-IVB three-stage boosts to earth orbit and the two-stage translunar injection boosts are proposed so that the maximum possible quantity of propellant loading for the Apollo service module, the Apollo lunar landing module, and the S-IVB may be realized. The possibility of providing excess velocity capability in all stages of booster operation, especially service module operation, will in general contribute to reliable margins-of-error allowances noted in Section II, Assumptions. In fact, these boost trajectories may be required so that the Apollo lunar landing mission may be successfully accomplished by the rendezvous of only two C-5 Saturn boost system payloads. It may also be required for additional staging for lunar orbit and landing. If this should be the case, the rendezvous equipment weight may increase. The proposed rendezvous method utilizes structures that are required for other reasons and does not penalize their design. The docking system requires a minimum of additional specialized components to complete the docking procedure.

RENDEZVOUS CONCEPT

The rendezvous concept described herein is restricted to the mating of large masses (approximately 100 tons each) in space or to the transfer of fluids between large containers in earth orbit. No method of transferring men or equipment between space vehicles is included in this phase of the earth-orbital rendezvous study. Specifically, the rendezvous and mating of Apollo lunar landing missions vehicles in earth orbit or the transfer of propellant from a tanker to Apollo boost modules is the main emphasis of this study.

The rendezvous concept includes placing an S-IVB booster or tanker into a 480-kilometer parking orbit. When adequate ground track of the S-IVB has been established, an Apollo lunar mission vehicle is placed in a low, 225-kilometer standby orbit. The differential orbit spacing of the Apollo lunar mission vehicle and the S-IVB booster permits in-space acquisition and tracking of the S-IVB by the Apollo at relatively low orbit velocities. When proper orbit angular relationships have been obtained, an elliptical transfer orbit of the Apollo lunar mission vehicle is made from the 225 kilometer orbit to the 480-kilometer orbit. The orbit transfer timing will permit the Apollo to be ahead and above the S-IVB at a slant range of approximately 10 nautical miles. The Apollo will be at a lower velocity than the S-IVB. Final orbit injection of the Apollo will be based on a radar range/velocity increment (range to go) in which radar and optical tracking inputs of the S-IVB target vary the Apollo transfer orbit into a station-keeping trajectory with the S-IVB. This form of automatic trajectory and velocity control is similar in system operation to an astronaut pilot manually performing the final station keeping maneuver. The pilot would note range, angles, and rates of change of these elements and, in turn, would command thrust magnitude and vector accordingly. When station keeping is established at 500 to 1000 feet, the rendezvous portion of the concept is achieved.

The Apollo lunar mission vehicle is equipped with a large conical docking bumper mounted on the landing legs of the lunar landing module. The S-IVB is equipped with an extendable boom located in the booster nose cone. The S-IVB boom is servo-controlled via beacon homing signals from the apex of the docking cone of the Apollo and is capable of a 15 degree, half-angle conical search pattern when extended. The docking and mating system, therefore, is an enlarged probe and drogue similar to an in-flight aircraft refueling operational mode. (See Drawing 7121-1901-2.)

The S-IVB has been solar oriented in a predetermined attitude while in parking orbit. Assuming the S-IVB longitudinal axis is parallel to the rays of the sun and that the booster nose is pointed toward the sun, the sun sensors of S-IVB are shadow null positioned on the instrument unit; and if the Apollo or the earth should shadow the S-IVB sun sensors, no new attitude reorientation of the S-IVB will occur.

The astronaut commands, by radio control, the S-IVB boom and servo systems to deploy with beacon receiver search systems active. The S-IVB telemetry will continuously transmit the boom deployment state and the angular deflections of the boom in relation to the booster reference axis. After confirming S-IVB docking readiness, the astronaut activates the vidicon cameras and the beacon transmitter located on the docking cone and maneuvers the Apollo into a coaxial orientation with the S-IVB. Sun sensors on board the Apollo command module direct the axial orientations of the Apollo maneuver, while the vidicon display indicates the axis offset of the Apollo and the S-IVB.

The rendezvous concept can be restated in the following manner: If the sun is utilized as a reference source by both the Apollo and the S-IVB, the longitudinal axes of the vehicles will form parallel vectors of the same pointing sense. The vidicon camera located on the docking cone will provide relative offset information of the vehicle axes. Further, the telemetry signals from the S-IVB regarding boom deflections will indicate the angular offset of the Apollo docking cone beacon and the angular deflection of the S-IVB docking-boom-tip-sphere-mounted receiver up to 15 degrees. When the boom is deflected 10 degrees or more, the astronaut will be able to view the S-IVB through the Apollo periscope system. When the angular offset exceeds 35 to 40 degrees, the S-IVB can be viewed directly through the Apollo windows.

The second astronaut command maneuver is to translate the Apollo vehicle until the S-IVB appears in the axial reference reticle of the vidicon/cathode ray screen. The S-IVB boom deflection angles will further confirm the vehicle coaxial alignment since zero boom deflection angles will be received in the Apollo.

After coaxial alignment has been established and maintained for a reasonable period of time, the astronaut can initiate an approach maneuver. Physical contact of the boom sphere to the docking cone may occur at a rate of closure up to 50 ft per sec. The sphere will skid along the cone and impact the automatic lock at the cone apex. The mass travel distance remaining after the boom is captured and locked by the docking cone is approximately 46.5 ft. The pure coaxial energy absorbtion by the boom is approximately 983,500 ft-lb at 100 psi or 9,835,000 ft-lb at 1000 psi. The boom is designed for a man-rated, 1000 psi operation.

are unlocked, and the gear is deployed to lunar landing attitude. The gear deployment releases attach fittings on the Apollo docking cone. The S-IVB and the rendezvous docking mechanism are thereby jettisoned; this results in an initial clearance of 5 feet between the Apollo and the S-IVB. The lunar landing module engines are fired to translunar cut-off velocity; and the rendezvous and docking sequence, required by the Apollo lunar mission objective, is thereby completed.

A second use for which the rendezvous and docking system can be utilized is the transfer of fuel from a tanker vehicle to an Apollo vehicle in space. All rendezvous and docking sequences are similar. However, after the boom absorbs docking impact energy, it is fully extended. The Apollo and S-IVB vehicles approximate a dumbbell configuration. The configuration is slowly rotated by combined application of the Apollo/S-IVB attitude control jets. The rotation settles the propellants in the two vehicles. The large mass centroid spacings develop adequate acceleration forces at low rpm. Further, the astronauts will always be located far enough from the axis of rotation to minimize motion sickness. For example, it was determined, during the NASA Self-Deploying Space Station study, that when a rotational radius of 75 feet exists, an astronaut can tolerate angular velocities of 4 to 6 rpm. In the configuration of Apollo/S-IVB depicted in Drawing 7121-1901-2, the astronauts' rotational radius will average over 80 ft.

For the fuel-transfer type of operation, the boom-tip sphere would contain a connect-disconnect valve, and the boom would perform the fuel transfer line function. Propellant, as a hydraulic medium, would extend the boom and power the search actuators. Pump power for propellant transfer could be derived from a propellant-gas-generator turbo-pump subsystem. All other systems would have similar requirements and hardware, except the master slave function of the S-IVB guidance for translunar boost, which would not be required. Attitude sense would be obtained by means of the sun-sensing systems on the Apollo and the S-IVB tanker.

COMPONENTS AND SYSTEM REQUIREMENTS

Only those items that are required for or that directly aid in the solution of earth-orbital rendezvous and docking are included in this section.

Ground Support Equipment

Precision Orbit Tracking

The limited payload capability of the Saturn C-5 boosts systems require, for Apollo lunar operations, that minimum energy boost trajectories be utilized for earth-orbital rendezvous operations. Although boosting the

Nominal relative approach velocities may be about 16 ft per sec, and the boom will have an operating safety factor of 10. Under nominal conditions, the booms will only be partially depressed to absorb collision energy. The astronaut will command boom retract, and the boom will now operate as a controlled tension tie between the Apollo and the S-IVB. As the boom retracts, the S-IVB nose cone, now acting as a bumper, will be drawn into the Apollo docking cone. At full boom retraction, the boom locks with preloaded tension; and the S-IVB is tightly socketed in the Apollo docking cone. Axial load and bending moment forces now existing between the Apollo and S-IVB are reacted by the cone socket action.

During the Apollo-to-S-IVB approach maneuver, the homing beacon receivers located in the S-IVB boom-tip sphere will continuously track the central area of the Apollo docking cone. The boom servo system will direct the boom-tip sphere toward the Apollo docking cone central axis. The approach may have an Apollo/S-IVB axis centerline offset of as much as 15 ft, and a successful boom capture will be made.

The angular difference of the nominal Apollo centerline to the approach vector of the Apollo can be as much as a 20-degree conical half angle as long as the offset of the boom sphere to the cone centerline does not exceed 5 ft. The S-IVB angular difference to the Apollo approach vector may be as much as a 10-degree conical half angle as long as the boom sphere is within 5 ft of the Apollo docking centerling. The nominal approach velocity of 16 ft per sec should be reduced if the above angular misalignments occur.

In summary, the gross conical half-angle difference between the Apollo centerline and the S-IVB centerline can be as much as 30 degrees with an additional 15-degree conical half-angle difference for boom axial compression to absorb the approach velocity energy. Maximum approach velocity energy can be absorbed if the vehicle axes are coaxially aligned. Attitude control forces are minimum when vehicle axes are coaxially aligned. The requirements of approach velocity versus angular differential of axes versus attitude control force form a complex differential force/motion dynamic relationship that mush be fully explored before adequate vehicle behavior or attitude control system requirements can be established. Until this research has been completed, the docking maneuver should be executed at low velocities along coaxially located vehicle axes or at small angles in such a way that force vectors pass close to vehicle mass centroids.

After docking is complete and the inertial platform of the instrument unit is slaved to the Apollo guidance unit, the combined mission assembly is oriented for translunar injection. On command from the astronaut or from the automatic computer, the S-IVB engines are fired to propellant burnout. Disconnects located on the lunar landing module landing gear pads

S-IVB into the 480-kilometer orbit does not require prelaunch tracking, the subsequent boost of the Apollo vehicle into the 225-kilometer orbit will. Precision tracking of an S-IVB in orbit before launching an Apollo vehicle permits considerably more Apollo payload in orbit in as much as fuel margins for angular and timing errors of boost trajectory may be minimized. S-IVB time in parking orbit will possible be shortened; the margins for propellant boil-off will be thereby reduced. In addition, the complete documentation of events will require precision tracking capabilities. Precision ground track of Apollo systems will be necessary in the foreseeable future and will contribute substantially to the Saturn C-5 payload-in-orbit capability.

High-Speed Telemetry

High speed telemetry can be utilized when cooperative, space-to-space and space-to-ground data are required. Ground stations will use telemetry for flight recording requirements. However, some telemetry information, such as vehicle attitude, remote platform erection, docking boom search angles, and operational-state data, can be utilized by the Apollo astronaut-pilot during docking maneuvers. In addition, orbit data confirmation on a continuous basis will greatly aid the astronauts in mission decision making. Continuous high-speed, telemetered data, properly displayed, will greatly aid in-space maneuvering. (Refer to Section V of this document.)

Computation Equipment

A nominal amount of computing equipment will be designed into the Apollo command module and the NASA Saturn instrument unit. A prohibitive vehicle weight penalty would result from all of the computer necessities being airborne. Earth-orbital rendezvous can be made easier by placing certain automatic computer equipment under the radio control of the Apollo spacecraft; such equipment could be used by the astronaut to solve special problems. In any event, such ground-based equipment should be available for back-up use by the Apollo astronaut/pilot.

Command Decision and Control

Normally the astronauts will be in full command of space operations. However, for safety (i.e., as a back-up to space operations), the rendezvous operations should be capable of being interrupted from the ground; this will also ensure the accomplishment of the safe return of the Apollo vehicle to some predetermined landing area. Further, some aspects of near-earth orbit-to-orbit changes of rendezvous might be more efficiently commanded from the ground. At this time it appears that deep-space orbit changes will be better performed if the astronaut/navigator observations and calculations are aided by ground-based computer functions of DSIF.

S-IVB Equipment

S-IVB Translunar Injections Booster Engine Requirements

It is recommended that any S-IVB utilized as an Apollo vehicle translunar booster should have engines whose thrust is equivalent to two Rocketdyne J-2 rocket engines. The 400,000 pounds of thrust provided by two J-2 engines provide the following:

1. Third-stage boost to 480-kilometer orbit with an initial T/W of approximately 1.32, as a self-injecting payload boost stage.

2. First stage translunar injection booster capability with an average, initial, estimated T/W of 0.82 and an average, final, estimated T/W of 1.43. In light of the many mission profiles associated with the Apollo project and related projects of deep space research, it is suggested that such performance capability may be near optimum.

Engine Restart Capability

To fulfill rendezvous operation objectives, the main thrust engines of the S-IVB should have multiple restart capability. The engines should by capable of multiple restart at anytime until the available propellants are completely exhausted. Such restart capability will require methods whereby the propellants are readily available for engine operation. Infinite engine restart may require a sizable propellant transfer storage system in which the propellants of the main tanks may be captured in a positive expulsion tank system during periods of main engine thrust. The positive expulsion tankage volume will be dependent upon the behavior of fluids in zero-g fields, system thermal properties, and the flow requirements of J-2 engine operation. Although no such transfer tank system for the S-IVB is shown on Drawing 7122-1901-2, such a system has been indicated on the lunar landing module in the drawing. It is recommended that an equivalent system be provided for the S-IVB and that the performance of such a system be included in S-IVB operational analysis studies.

Control of the main S-IVB engine thrust vector is mandatory. Third-stage earth-orbit, self-injection operations and first-stage translunar, boost-injection operations require thrust vector control. Precise thrust vector control with adequate rates of response will materially reduce the quantity of propellants reserved for error margins.

Remote Erection Capabilities of the NASA Instrument Unit Guidance Platform

The NASA instrument unit directs the electronic guidance and control functions of all Saturn boost systems except Apollo. Included in the unit are telemetry systems, command radio tranceiver systems, and a control command signal system in which all boost phase stage operational signals are generated. No attempt will be made here to describe all of the instrument operational capabilities.

One requirement recommended for inclusion in the functional capability of the NASA instrument unit is the remote (Tanlock-type) radio-link master/slave erection of the guidance reference platform. It is impractical to mechanically mate the S-IVB to Apollo, by an earth-orbital rendezvous operation, so that the thrust vector of the S-IVB is precisionally aligned to the Apollo vehicle. The required precision of axial and roll-orientation alignment of two large vehicles may be greater than can be accomplished through mechanical mating. Even if such a precision mate were possible, the length of the assembled vehicle (approximately 185 feet) would be subject to curvature changes in the combined mechanical axis due to temperature change and other differentials. (The vehicle side facing the sun would tend to expand, while the other side would tend to contract.) The resulting curvature may cause S-IVB thrust vector alignment to be in error to Apollo command vectors and therefore beyond mission precision requirements. The S-IVB engine control connected directly to the Apollo guidance system would place undue size and complexity requirements on the Apollo guidance system. The S-IVB control system would become mission sensitive and thereby require complex modal capacities. The remote mating of electrical connectors whose mating tolerances are usually of the order of 0.001 to 0.005 in. would be extremely difficult. Many more examples could be given, but those stated are sufficient to explain the master/slave relationships.

The radio-link would be utilized to effect coincidence alignment of the Apollo and S-IVB instrument unit reference platforms. The S-IVB instrument unit guidance platform is normally tied to the engine vector control system; and when slaved to the Apollo platform, it acts as a resolver unit for Apollo command sequences. The thrust vector of the S-IVB is effectively controlled by Apollo commands and trajectory responses. The S-IVB structure does not require any angular indexing about the longitudinal axis relative to the Apollo vehicle longitudinal axis. The requirement for precision mating of electrical connectors is eliminated. Further, the NASA instrument unit can now be erected on the launch pad without the use of umbilicals. The launch vehicle size may be so great that long-wire-loss input will degrade the accuracy of guidance erection. Also, both the Apollo and S-IVB platforms could be remotely erected by ground stations whose trajectory constants include precision observations smoothed by computer networks into extremely precise guidance reference data. Complex control mode systems become ground based, and only mission mode sequence control equipment is needed as flight hardware.

Solar Orientation Sensing and Vehicle Attitude Control System

Solar orientation sensing is utilized for the attitude control reference of the S-IVB when it is in the 480-km parking orbit. Over protracted time periods, the NASA unit guidance platform would be unable to maintain the proper S-IVB orientation to the sun that may be required by propellant boil-off control. The solar orientation system can maintain proper orientation as long as attitude control propellant is available. Either the solar orientation system or the guidance platform of the S-IVB can be utilized during rendezvous and mating; however, there is less electrical power required by the solar sensing system in comparison to platform electrical consumption. The telemetered output of the S-IVB sun sensors are adequate for docking maneuvers and boom deflection displays.

Tracking Beacons

The Apollo vehicle and the S-IVB should be equipped with tracking beacons. Two types are recommended. The first is a radar transponder beacon that will reinforce radar target return with IFF-type coding of the amplified return signal. The second beacon is carrier wave (CW) in nature and permits the rapid acquisition of the target for tracking. Normally such beacons are installed on each stage of the launch vehicle and its payload so that proper identification of launched systems can be made. Such tracking aids are necessary for earth-orbital rendezvous if the Apollo is to acquire, track, and home on the proper target during the transfer orbit maneuver. Further, the reinforced signal strength of the S-IVB improves the quality of target returns by several orders of magnitude and thereby significantly reduces the guidance allowance-for-error margins required when no beacon system is employed. If the Apollo picks an incorrect target on which to home during the orbit transfer phase of earth-orbital rendezvous, the Apollo lunar landing mission becomes impossible.

High-Speed Telemetry

Most major subsystems of a C-5 booster and its payload will be equipped with telemetry facilities. In general, the telemetered information is supposed to be picked up by ground stations for recording or for remote command control functions. If the Apollo vehicle is equipped to receive the telemetered data of the NASA instrument unit and the S-IVB booster and to properly display this data, the earth-orbital rendezvous operation will be vastly simplified. Not only can direct electrical mating of Apollo/S-IVB be avoided, but also in-space checkout of the S-IVB by the Apollo astronauts can be accomplished; and, as previously stated, electrical alignment of the vector sense of S-IVB thrust is more accurate than mechanical structural mating. Cold welding of metal parts in a vacuum may require radio-link connections of control signal circuits in place of mechanically mated umbilicals. The docking manuever is simplified by the proper display in Apollo of the docking boom functions.

Apollo Equipment

Vehicle Configuration

The vehicle configuration controls, in detail, the specific components used for docking in space. For instance if the Apollo command module structure were strong enough to transmit S-IVB booster loads during translunar injections and the command module air lock and docking provisions for the space laboratory could absorb the velocity energy of S-IVB docking, only the S-IVB would require the installation of docking provisions. If, however, the damage sensitivity of the command module exterior is high, then such a docking system cannot be used. Multiple staging of the lunar landing module, such as a lunar propulsion module equipped to retro to lunar orbit only and a landing service module equipped with impact absorption gear, would require that both the Apollo and the S-IVB be equipped with docking provisions.

The docking provisions shown on Drawing 7121-1901-2 follow the concept of maximum vehicle spacing at initial docking contact with controlled closure during the final portions of the docking maneuver. All modules are considered to be damage-sensitive; they are protected by the docking equipment and the maneuver procedures. Structural misalignment is assumed to be small but not precision in nature when the structures are fully mated. Misalignments are compensated by the master/slave relationships of the Apollo/NASA instrument unit guidance platforms. The precision mating of umbilicals should be avoided, and structural indexing about the longitudinal axes of Apollo and S-IVB is not required. All provisions for docking are jettisoned when they are no longer required.

The lunar landing module landing gear is utilized as a common structural interface. The gear can take S-IVB booster loads without penalty, and it acts as the support for the Apollo docking cone equipment.

Coherently Coordinated Attitude Control Systems

More studies have delved into the multiple staging of the vehicle in the vicinity of the moon, and the Apollo vehicle has undergone many revisions. One such configuration iteration involves the following modules:

1. Apollo command module-mission unchanged

2. Apollo service module-propulsive capability increased both in thrust magnitude and duration.

3. A landing service module-mission to de-orbit from lunar orbit and land the command module, service module and landing service module. (The landing service module is equipped with landing gear.)

4. A lunar propulsion module mission to establish a lunar orbit from a translunar orbit for the command module, service module, landing service module and lunar propulsion module. The lunar propulsion module also supplies the translunar midcourse velocity corrections. The lunar propulsion module is jettisoned prior to landing service module de-orbit and landing function.

If we further iterate and propose additional functions for the lunar propulsion module to perform (1) second-stage translunar injection boost, (2) 225-km to 480-km orbit transfer boost, and (3) third-stage boost to earth orbit as a self-injecting Apollo/C-5 system payload, we may obtain the following potential advantages even though certain degradation of the lunar propulsion module performance exists:

1. Increased payload in earth orbit

2. Minimum equipment for orbit transfer operations

3. Increased payload in the lunar orbit

4. Increased payload on the moon

5. Reserve velocity increment during earth return of the Apollo command module to retro (in part) to earth orbit from the transearth orbit

The trajectory iterations are made to emphasize the need for all modules under the direct control of the Apollo command module (by the astronauts or by the automatic equipment) to have attitude control systems that can function properly without multimode controls. The astronaut should only prescribe the vector sense and response rate he expects of the attitude controls with the same control units. He should not be distracted by mode switching problems. Therefore, as each booster module is jettisoned, the astronaut should be able to provide the necessary attitude control over the remaining modules without sequence switching. This can be possible only if all attitude control requirements are considered in a single overall control system solution. The result will be a conerently coordinated attitude control system.

Optical and Electro-Optical Display System

The Apollo man/machine complex can perform many functions not assignable to machines alone. This is especially true of the final phases of rendezvous, such as station keeping and docking. The astronaut-pilot can perform almost any maneuver he chooses with the Apollo if he can see the elements of the problem. Optical and electro-optical aids that extend the astronaut's visual range or that improve the accuracy of observations become essential in space operations. The syncronization of vehicle orbits is greatly simplified if telescopes extend visual range. When the telescope image is electro-optically superimposed on a situation display wherein all tracking elements are in proper relationship, the space maneuver of rendezvous can be efficiently performed by the pilot. Station keeping and docking maneuvers can be performed efficiently on a direct, visual flight procedure. The blind spots caused by vehicle shape dictate the use of vidicon camera aids for docking as proposed. Telemetered system functions appearing on cathode tube displays improve docking efficiency and accuracy. The procedures and maneuvers previously described underscore the need for an optical and electro-optical display system. Some of the system elements are the following:

1. Observation window, required for direct visual functions

2. Combined periscope/telescope, required for visual range extension and tracking input parameters for electro-optical situation display

3. Vidicon or image-orthicon cameras and a cathode-ray tube display, strageticall located to eliminate astronaut blind spots caused by vehicle shape

4. Cathode-ray tube data display, required for conversion of telemetered data into situation display images for improvement of maneuver accuracies or rates of maneuver changes (The radar system also provides essential inputs to the situation display system.)

In-Space Acquisition and Tracking System

New requirements for acquisition and tracking of objects in space and for determining the orbit relationship of such objects relative to Apollo evolve from the man/machine complex of Apollo. The earth-orbital rendezvous of two or more vehicles may prove to be more efficiently accomplished by on-board acquisition and tracking elements. As the information path time lapse increases, it is more apparent that this assumption is true. Ground-based tracking may be capable of adequately defining navigational parameters. However, when two objects remote to ground tracking facilities are to be mated, local events will occur faster than ground track can supply essential data. Orbit transfer data may be acquired and smoothed by ground based equipment to a high degree of accuracy, but the actual performance of an orbit transfer and rendezvous maneuver will be more accurately performed by a space-borne system. Whenever station-keeping relative velocities or ranges become small, the precise determination of magnitudes is beyond the capabilities of ground-based equipment. However, short range Apollo systems would more than adequately define the magnitudes involved. The stringent safety requirements imposed by the presence of men in the operation of space rendezvous underscore the need for precision equipment to be carried by Apollo.

The philosophy of redundant systems and alternate system-usage will, in general, apply to the acquisition and tracking requirements of the Apollo vehicle systems. Some equipment components that will probably be required are the following:

1. Inertial platform - local reference for relative measurements
2. Radar
    a. Long range for acquisition and tracking
    b. Short range for small distance measurements
    c. Doppler for small velocity increment measurements
3. Computor/guidance - problem solving of orbit parameters and command requirements
4. Pilot displays
    a. Cathode-ray tube for multimode display requirements
    b. Meters for smoothed rate function and range displays
5. Optical
    a. Windows - direct visual for pilot usage
    b. Telescopes - long range visual

c. Electro/optical system, for passive tracking and blind spot elimination

d. Periscope, to permit externally located optics to present information internally on display screens

e. Infrared, for back-up sensing and tracking system

6. Beacons

a. Carrier wave, located on target for rapid acquisition

b. Transponder/radar, located on target for rapid identification of target on radar display

The utilization and description of the system elements have been discussed previously.

Multimode Guidance Systems

The Apollo vehicle guidance systems used for earth-orbital rendezvous will probably be separated into several different control modes. The number of modes will depend upon whether or not a NASA instrument unit guidance or an Apollo guidance will control the Saturn C-5 boost system. It is currently contemplated by NASA to use an instrument unit for all C-5 boost guidance and to use Apollo guidance after the S-IB, S-IIB, and/or S-IVB have been expended. The Apollo guidance still has many unsolved problems, not the least of which are the orbit transfer from 225 km to 480 km and the rendezvous with possibly an S-IVB booster. Some Apollo missions require rendezvous and docking with earth-orbiting space laboratories and space stations. Specific rendezvous procedures are not yet determined, but the following maneuvers will require different modes of guidance and control.

1. Launch guidance of C-5 Saturn. The excess weight of a NASA instrument unit may potentially be saved. Depending on the additional complexity to the Apollo guidance system, this type of guidance may be an inertial type or radar-beam-rider type.

2. Orbit transfer. This operation will probably be performed with a Saint vehicle rendezvous station-keeping maneuver. Long-range and short-range radar provide range, velocity, and path angle data for a homing, intercept type of guidance control. Since the power required to operate optical sensors is less than radar and since no atmospheric barriers exist in space, SOLO and HALO type electro-optical trackers will probably aid radar for transfer orbit operations.

3. Docking. Docking requires a special type of guidance input. Docking can be manually performed by a pilot, and analog simulators show that certain approach and tumbling maneuvers can be performed with minimal rate and position data. Therefore, the guidance mode may be a manual or a taped program.

4. Translunar injection. This flight maneuver and boost function will probably include remote control from ground facilities, astrotracker/inertial guidance (as set up by the astronaut celestial observations), and manual guidance (as may be predicated by optical star sightings and the earth/moon/Apollo angular relationships).

The possible guidance modes are the following:

1. Inertial/preprogramed

2. Radar beam rider

3. Radar/inertial homing

4. Electro-optical/inertial homing

5. Stellar/inertial

6. Manual

7. Remote radio/telemetry/inertial

The trajectories and maneuvers previously described for rendezvous and docking of the vehicles shown on Drawing 7121-1901-2 will probably include guidance systems 1, 3, 4, and 6 above.

Propulsion Systems

The variety of trajectory, maneuver, and docking operations described herein indicate the need for thrust magnitude and vector control over a wide range of values. As previously stated, the lunar propulsion module may be called upon to provide power for many phases of the lunar flight, including orbit transfer, rendezvous, and docking. Orbit transfer tends to require high thrust combined with precise vector control of thrust, applied at least twice. The first powered phase initiates orbit transfer. The Apollo then coasts to the rendezvous transfer point. At the transfer point, synchronization with orbit ephemerae, is achieved through a second burst of thrust. This synchronization orbit plane changes through thrust and vector control. Impulsive thrust is utilized for the transfer operation and will probably be a very high thrust of short duration. Station keeping and docking will probably be accomplished at low thrust and at increased time intervals in a manner that will reduce relative vehicle velocities and range spacing to zero. Final docking will probably utilize the very low thrust of the attitude control systems. For reasons of safety, the thrust will probably be throttleable so that the astronaut-pilot may complete the maneuvers required with maximum control and, therefore, maximum safety. Slow boom retraction in the order of 0.5 to 2.0 feet per minute will probably characterize final vehicle closure even though, for design assumptions, much larger values of relative velocity were used. Safety and damage sensitivity dictated the high margin of safety for component design. Gemini will reveal many of the problems and solutions to the problems associated with rendezvous and docking. Specific inputs to Apollo will certainly modify any concepts preferred at this time.

Docking Equipment

The docking equipment design is based on the maximum utilization of the existing structures of the Apollo and S-IVB vehicles. The docking system is based on an enlarged probe and drogue currently utilized by airborne refueling techniques. The system has been designed for very high margins of safety in as much as inadvertent collision in space of two 100 ton vehicles would result in catastrophy. The specific concepts of the system are explained in the description of components.

S-IVB Booster Equipment Groups

The S-IVB has been assumed to be a translunar injection booster for Apollo. The aerodynamic nose cone has been redesigned to act as a bumper; this results in a minimum increase in weight and in a structure with low damage sensitivity. An extendable probe equipped with servor system for cone housing is internally installed. The S-IVB is solar oriented; and during rendezvous and docking, no maneuvers by the S-IVB are contemplated. The S-IVB attitude control system, however, must be powerful enough to prevent pitching or spinning motions when the probe impacts the Apollo docking cone. The probe boom and nose cone have been designed to accept miss impact forces in such a way that additional docking passes by the Apollo vehicle are possible.

The NASA instrument utilized for boost guidance is assumed to be capable of acting as a slaved resolver guidance unit subject to remote radio platform alignment and command control. See Drawing 7121-1901-2 for details of docking equipment components.

Docking Boom and Pivot Actuator System. The docking probe boom is a multiple extending pneumatic cylinder strut. There are six extendable sections and one structurally fixed, gimbaled section. The innermost cylinder is approximately 6 inches in diameter and 112 inches in length. The outer cylinder is approximately 23 inches in diameter and 110 inches in length. The five remaining cylinders form pistons whose annulus area is approximately equal to the area of the innermost cylinder. The cylinder assembly has been designed for a maximum operating pressure of 1000 psi. The pneumatic plumbing is capable of slow fill rates to extend the boom, which is by-passed by high flow pressurization and blow down plumbing. The boom is also equipped with a length change sensor system. As the impact energy is absorbed and the boom compression length rate change approaches zero, the compressed gas that extends the cylinder is by-passed, by automatic controls, to the retract cavities of the sleeves until the pressure is equalized. This prevents vehicle rebound forces that result from the use of pneumatic power becoming active. Excess pressure is also vented from the boom via a pressure regulation system.

Radio commands from Apollo control, extend, and retract the boom. When connected to Apollo, the boom sleeves are safety locked under preload to maintain the socket mate of the Apollo docking cone to the S-IVB nose cone.

The outer boom cylinder is gimbal mounted to the forward end of the S-IVB nose cone. The aft end of the boom is connected by two actuator cylinders to the S-IVB nose cone base. One cylinder is located in each of the orthogonal reference planes of the S-IVB that pass through the S-IVB longitudinal axis.

A beacon homing receiving system is located in the boom-tip-sphere. Signals eminating from the apex of the Apollo docking cone cause the actuators to pivot the boom toward the cone apex up to a 15-degree conical half-angle to the S-IVB longitudinal axis. The actuators may be electrically, hydraulically, or pneumatically powered, according to the power sources available on the S-IVB. The boom length, angular offset, and pressure are telemetered. The telemetry signals are utilized by the Apollo astronaut utilizes the telemetered signals, displayed on data display equipment, to assist in the docking maneuver.

The design of a specific boom must consider energy rebound, extension-rate control, column and beam-column boom strength, sliding seals, cold welding of metals, meteorite protection, homing quality of beacon system, search rates, damage sensitivity, and many other items that are associated with space environment.

Bumper Structure. The nose cone of the S-IVB functions as a bumper for rendezvous docking. Normally the cone structure is strong enough to resist the aerodynamic forces and thermal stresses during boost. The redesign of the cone into a bumper structure results in very little increase in weight. During launch, aerodynamic forces of approximately 100,000 lb. are resisted by the cone on a distributed force basis. Approximately two to three times the aerodynamic force can be resisted by a cone of the same weight, if such force is applied on the cone apex through a properly designed fitting. The additional weight, therefore, results from making the cone insensitive to impact with the Apollo docking cone.

The structure will probably be fabricated of honeycomb or corrugated, reenforced sandwich structure with an outer skin that is thick. Integrally, Chem-milled stiffeners and rings on a rolled aluminum plate would also provide damage insensitivity during docking.

The NASA instrument unit shell and the forward end of the S-IVB will require that docking forces be considered during design.

Boom Beacon/Homing Receiving System. The boom beacon/homing receiving system may be UHF radio, infrared, optical, or visual-light electro/optical. Each type of homing device can be signal coded to reject all signals other than those emanating from the Apollo docking cone apex. An electro/optical system of the SOLO seeker type will, in all probability, be the smallest, lightest system and will require the least power to operate. A high-intensity, pulse-modulated, monochromatic light source located on the Apollo docking cone apex would be extremely easy to detect and discriminate from other light or energy sources. Equipped with a precision light light filter, the SOLO optical system would reject stray light. When fed into a decoder detector, the light-sensor-pulsed output would further identify the light source as the proper target.

In space, a monochromatic, pulsed, high-intensity light source could be detected several hundred miles by the small (approximately fifty millimeters) optical head. The SOLO unit is also capable of 0.1-degree angular accuracies, as an inherent reliable design capability. The capability of SOLO is more than adequate in range and in accuracy for docking.

The electro/optical function of SOLO effects pointing of the detector head. Pickoffs located between the head and boom support define the angles for boom search motion. The pickoff signals are converted to boom actuator command signals, and they pivot the boom until the SOLO head angles are zero. The boom is then co-axial to the head.

Boom actuator position signals are fed into both SOLO and telemetry circuits. The SOLO unit utilizes the signals to prevent excess boom pivoting. At boom pivot angles of approximately ten degrees, the search pattern may be stopped, and the attitude control system of the S-IVB could realign the S-IVB axis toward the Apollo docking cone. Normally such S-IVB realignment is to be avoided since the Apollo and the S-IVB utilize the sun as a common reference point for the docking maneuver. However, the S-IVB sun sensors could display such angular changes via telemetry on the Apollo display screens. When sun and boom angles become zero, the previously described docking sequence will have been established.

Boom Command Receiving System. The NASA instrument unit could receive and convert Apollo command signals into boom operating commands. The boom actuation (extension, retraction, and search) would require signal conditioners and servo circuits to finalize the Apollo command. If the NASA unit is not so equipped, a command-receive system for docking would be required; this could be located in the nose cone cavity of the S-IVB. Through proper pulse coding of the Apollo light source, the SOLO unit could act as a command-control and a boom-tracking system. There are many alternate transmitting and receiving systems available. The basic question concerns the reduction in number of new components and their reliability from an inherent property basis.

Pneumatic Power System. The docking boom requires a regulated gas supply of considerable volume. For initial extension of the boom, a low pressure source of gas is required. This source might be prepressurization of the boom by trapped ambient gas (clean, dry N_2 at one atmosphere) used to purge the boom prior to launch. After the boom has been extended in space, a high pressure gas should be injected into the boom in such a way that docking energy can be absorbed. During boom retraction, pressure depends upon the rate of vehicle closure.

A solid propellant gas generator could supply the high-pressure gas requirement. Electrical heating and boiling of a super-critical cryogenic gas supply could also furnish the necessary gas volumes. The pressurization system of S-IVB could also be a source of pressurized gas. The requirement of a supply is only noted here, not defined.

System weight dictated the use of pneumatic actuation of the boom axial motion rather than hydraulic action. Weight and problems of lubricating and cold welding metals in space eliminated electrical devices for boom extensions. Safety dictated a strong structure for the boom in lieu of the preformed strip-metal ribbon tubing proposed by American Machine & Foundry. Concentricity of boom location was chosen to eliminate force eccentricities as much as possible. To eliminate cold weld problems, the metal sleeves of the boom ride on teflon seals and slipper rings. High pressure capability permits the absorption of high-impact energy, which renders the problem of regulating approach velocities less sensitive.

Boom Operational State Telemetry System. This equipment and its operation has already been described in this rendezvous and docking discussion. This paragraph is introduced to account for the system under S-IVB components and to underscore the fact that the telemetry of the boom systems will be invaluable during docking. Telemetry, a normal function of space flight data transmission, becomes a powerful tool during docking; and an attempt will be made to express the possibility, in practical terms, of a concept or approach to the problem that may eliminate such requirements. The utilization of the NASA instrument unit in a new mode of control is a major key to such requirement elimination.

The NASA instrument unit, in conjunction with launch control commands, generates all signals for launch guidance, vehicle attitude-control systems function, sequencing, staging, thrust vectoring, jettisoning, etc., of the Saturn boost systems for all stacks and combinations of booster and payload modules from launch to payload orbiting.

The final method of establishing the inertial reference of the NASA instrument unit has not yet been determined. Because of the physical size of the Saturn system, a very accurate ground-based master platform will electrically or electro/optically erect the NASA instrument unit by a remote control procedure. Signal errors caused by long cabling and physical deflections of the launch tower caused solar heat and wind vibration will render direct umbilical erection procedures inadequate. If remote erection of the inertial platform is used, it may prove a superior method of vehicle alignment and docking control when compared to other systems.

Platform Erection of the NASA Instrument Unit Guidance Platform By Radio-Link Techniques. Probably the most significant facts produced by this study are the potentials in the radio-link control of the NASA Instrument Unit Guidance System. Heretofore, every company that has studied the mating of translunar or deep space injection boosters to Apollo type systems have required the following:

1. Three-axis mechanical coincidence of structures when mated, to permit proper thrust vector control accuracies required by injection boosting

2. Mating of umbilicals of all kinds and types, so that direct control over the booster systems is exercised

3. Complex navigational and docking aids that produce little toward the real solutions of the problem

4. Extremely slow docking maneuvers dictated by so-called damage-sensitive connections

5. Precise alignment of mated parts or flotation of components to avoid mating problems (This sometimes introduces new problems.)

There are many other requirements, usually specific to docking procedures, too numerous to mention here. It is only fair to state that the facts developed by other studies are valid and require adequate consideration for real system application. However, the concepts presented in this document inherently eliminate most of the aforementioned obstacles and requirements. Typical examples are the following:

1. Three-axis mechanical coincidence is eliminated. Though the Apollo and S-IVB longitudinal axes should be reasonably aligned to minimize structural loading, roll alignment is unnecessary for purposes of guidance.

2. Since the NASA unit controlled all Saturn functions during boost and orbit injection, no umbilical connections of any kind are required. Telemetry profides the information and checkout data of system readiness. Master/slave inertial platform coincidence satisfies all boost injection guidance and control requirements.

3. Telemetry signals combined with electro-optical systems (used during lunar landing) can produce adequate docking displays.

4. Large, coarsely aligned, docking provisions designed to be impact insensitive permits wider approach velocity margins.

5. Precise alignment of mated parts is totally eliminated.

Vector coincidence of Apollo and S-IVB inertial platforms is the key and remote radio-link erections of the S-IVB inertial platform to the Apollo platform is the means to the elimination of burdensom requirements and to the solution many of the aforementioned space docking problems. (See previous discussions of Section IV for further information.) Existing equipment can be fully applied because of the mission capability currently being designed into the NASA instrument unit.

Apollo Vehicle Equipment Group

In general, the Apollo equipment group used for earth-orbital rendezvous and docking is a conterpart of the S-IVB equipment group. The structure of the Apollo modules was analyzed to determine whether any of the modules contained components that could be utilized directly for docking or for providing the support for special docking equipment. One such component, the landing gear of the lunar landing module, proved capable of reacting lunar landing forces, docking forces, and S-IVB translunar boost injection forces. Two minor functions that would not normally exist were given to the landing gear. Docking cone support and release fittings were added to the landing shoes, and the shock struts were changed to re-usable designs, of the air-oleo type. The Apollo pilot display system, a second group of components, provides docking system data displays with the addition of a small telemetry receiver. S-IVB guidance and vector control via remote radio link eliminates the need of penetrating the Apollo shell with additional umbilical control lines and wiring. The major objective for the Apollo group of docking equipment is the maximum utilization of existing systems with the least possible amount of new equipment being required.

Docking Cone. The docking cone proposed for earth-orbital rendezvous docking is similar in structure to the previously discussed S-IVB nose cone structure. The cone base is approximately 170 inches in diameter and has a 30-degree half-angle. The cone near base, for approximately 10 inches, is additionally inclined to a 45 degree half-angle to clear the installation of a torodial expulsion bag ring. The expulsion bag absorbs final docking impact forces while deflating and permits the S-IVB nose cone to intimately contact the Apollo docking cone after deflation.

Equally spaced around the cone base are four attach fittings that receive the LLM landing gear attach fittings. A fail-safe sider lock that receives the S-IVB boom engagement sphere is mounted at the apex of the cone. The slope of the grooves and slider faces permits ready entrance of the boom sphere into the lock; this, however, prevents escape of the sphere after entry to the lock. The lock also contains spring-loaded sleeves, reenforced with pneumatic action, that can retain the sphere under all expected loading conditions. A beacon transmitter mounted on sliders is centrally located within the lock cavity. As the S-IVB engagement sphere enters the lock, the beacon unit is expelled from the lock and is automatically shut off.

In addition to supporting the docking cone, the lunar landing module landing gear provides additional energy absorption capacity in the event that final docking velocities are excessive. The gear design suitable for lunar landing provides more than sufficient structural strength for docking or S-IVB injection boosting. Therefore, only the new cone attach/disconnect fittings added to the gear landing shoes constitute additional weight assignable to the docking system for its support. The entire docking system is jettisoned after S-IVB burnout.

Boom Beacon-Homing Transmitter. The types of beacon homing systems available were discussed previously and will not be repeated here.

The beacon transmitter was located in the cone apex for several reasons. The initial contact of the docking boom to the docking cone will tend to be on the Apollo centerline and will result in lower disturbing forces to Apollo. Secondly, the beacon beam width constrained by the cone prevents the boom from homing on the surrounding damages-sensitive structure. Also, cockpit displays of the S-IVB docking system telemetry provide additional information redundancy for the astronaut. The axial location of the beacon helps define vehicle coaxial alignment during docking.

Radio-link for the Erection of the NASA Instrument Unit Platform. The type of tranceiver system that will probably be required for remote erection of the NASA instrument unit inertial platform is the Tanlock system developed by NAA; although in its present form, the Tanlock system will require modification. The principles embodied will contribute to high-quality remote platform alignment. The Tanlock system provides essentially all the elements of closed-loop communications control, including compensation for relative velocities between vehicles utilizing the system. The closed-loop operation is obtained via the tranceived signals and results in high-quality signal-to-noise ratios and signal resolution. The system can more than equal the signal quality required to erect guidance platforms that are currently umbilically connected to ground alignment equipment. Precise knowledge of the system and its functional capability will require review of the radio-link erection concept by those S&ID personnel involved in Tanlock system development. It is suggested that only scaling amplifiers and buffers designed specifically to match the inertial platform requirements will constitute the limit of additional components to be added to the Apollo Tanlock communication system that is currently under development.

IV. GUIDANCE AND CONTROL STUDY

CONCEPT

Introduction

The rendezvous mission is characterized by an orbiting Saturn S-IVB tanker vehicle and the maneuverable Apollo in near-earth orbit. Rendezvous is accomplished by transferring the Apollo vehicle from its established orbit to the Saturn orbit. The guidance problem consists of successfully directing the Apollo in a transfer path, which effects, ultimately, physical contact with the Saturn vehicle. The Apollo must achieve a condition wherein its position and velocity vector match the corresponding characteristics of the Saturn.

If adequate knowledge of the orbital characteristics of the S-IVB tanker and the relative position and velocity of the Apollo vehicle are assumed, various guidance techniques can be employed to perform the initial phases of the transfer. (See Figure 1 in the supplement to this report, SID 62-834-2.) It is not the purpose of this study to consider in detail the requirements and methods concerning this phase, but rather to specify concept and hardware associated with the terminal phase of rendezvous. Terminal maneuvers require accurate knowledge of relative position and velocity conditions, and additional data must be obtained through the use of vehicle-borne sensors to measure relative quantities directly. Velocity and position errors resulting from initial transfer guidance can be nulled out by using terminally acquired data in a succession of vernier-type maneuvers to match the position and velocity of the two vehicles.

In general, the rendezvous concept involves transferring the Apollo from its orbit to a path that is ultimately constrained to a near collision course with the nonmaneuverable S-IVB vehicle. Efficient rendezvous presupposes target position predictability. The Apollo orbit will be established at approximately 100 nautical miles, and a near-Hohmann transfer will be executed with the S-IVB in an approximately 400-nautical-mile orbit. The intercept vehicle (Apollo) will undergo a transfer orbit injection, coast phase, midcourse maneuver, and terminal or homing maneuver to couple the two vehicles. The use of a near-Hohmann transfer and the deterioration of target position predictability with time (the transfer orbit being determined from position data available at injection) probably force the requirement of a midcourse correction maneuver to effect a satisfactory rendezvous. Essentially, the Apollo guidance system must determine a course of action based on the predicted time of the tanker's arrival at a specified position to reach the target position at target arrival time. Actual rendezvous is repeatedly del ed until the line-of-sight rate is nulled and the relative range between vehicles is small. A negative range rate is maintained throughout the rendezvous manueuver.

Transfer Injection

The guidance objective for transfer orbit injection is to place the intercept vehicle in a position suitable for efficient midcourse corrections. This is accomplished by placing the Apollo on a trajectory that would closely approach rendezvous conditions without considering errors in target position and velocity prediction, injection, and transfer guidance. Injection guidance is the determination of the required velocity vector for the intercept vehicle to arrive at the selected rendezvous point at target arrival time. Guidance computation includes consideration of predicted time of arrival of the target at a selected intercept point, intercept vehicle position, and intercept vehicle velocity. A velocity-to-be-gained vector is computed for steering and thrust cutoff. Assuming rapid and accurate processing of the required data, precise control may be exercised during the entire powered phase of injection. In the absence of injection errors, the vehicle, subquent to cutoff, would pass through the determined intercept point at the required time. Mechanizations for solution of the guidance problem will not be considered. The required data for computer solution of the problem is readily available within the present state of the art, i.e., in an inertial system, radio command, or a combination of both. Accuracy of subsequent vehicle transfer motion, compared with desired or predicted motion, will depend upon propulsion adequacy, measurement accuracy, correctness of guidance mechanization, thrust control precision, and reliability of operation.

Correction Manuever

Midcourse guidance will be used primarily to correct the transfer orbit for injection guidance error. Target prediction errors may also be corrected if more up-to-date information is available. A correction sufficient to ensure a high probabiltiy of target acquisition is required. Relative vehicle positions must allow Apollo sensor acquisition to complete the terminal phases of rendezvous.

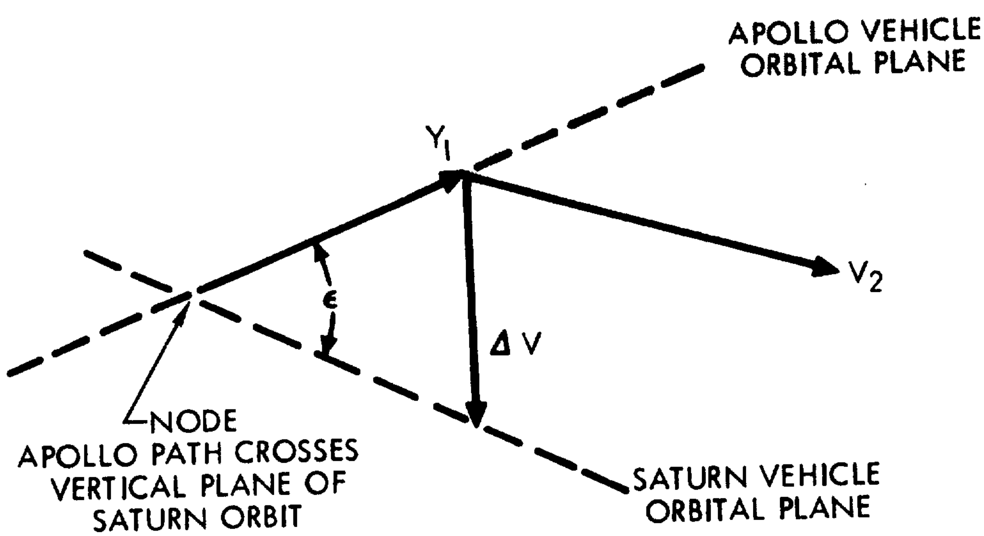
The magnitude of a midcourse maneuver must be kept as small as possible, because errors in the vicinity of the target will propagate proportionally to the magnitude of the required correction and because propellant requirements will obviously increase with the size of the maneuver. Major midcourse correction guidance considerations are the following:

1. The intercept vehicle must approach the target vehicle within a nominal miss distance consistent with sensor detection range.

2. The probability of target detection and the fuel expenditure required during the terminal phase will effect the ultimate allowable closing velocity.

3. Approach geometry and time required to successfully complete the terminal phase will affect the mid-course correction chosen.

The correction maneuver should place the Apollo vehicle in an orbit very nearly that of the S-IVB. Any out-of-plane (S-IVB orbital plane) velocity components that may exist should be reduced so that the transfer path and subsequent intercept vehicle orbit are in the orbital plane of the target vehicle. It should be emphasized that a possible 3.5-degree inclination may initially exist between the S-IVB and Apollo orbits. The initial plane change is accomplished when the Apollo orbit is being established, and transfer is initiated when the two orbits are very nearly coplanar. Optimum time for initiating the plane change from a propellant standpoint occurs at the intersection of the two orbits (line of nodes) as shown in the following sketch:



$$\Delta V = \sqrt{V_1^2 + V_2^2 - 2V_1V_2 \cos \epsilon}$$

where

V_1 = Initial velocity

V_2 = Final Velocity

ΔV = Required impulse for velocity vector change

The required velocity impulse (thrust vector) is oriented perpendicularly to a bisector of angle E. Any other thrust vector orientation will also cause a similar change in flight direction; however, a change in vehicle velocity will also result that is costly in terms of propellant requirements. Thus, for maximum plane change efficiency, $V_1 = V_2$; and it is seen that

$$\Delta V = 2V_1 \sin\frac{\epsilon}{2}$$

The correction guidance problem becomes one of determining the time for maneuver initiation and thrust vector orientation compatible with the foregoing requirements. Accuracy at the target will be affected generally by two error sources: determination of relative vehicle positions and execution of the correction maneuver. Accurate determination of orbit and position and subsequent prediction of future conditions depend upon error introduced by tracking data, including knowledge of site location propagation astronomical unit, etc., and its subsequent use in computations that may not be precise. Typical tracking capabilities of presently available equipment are presented in Table 4-1.

Present state of the art indicates that techniques are availabe to produce angular accuracies of a few seconds of arc and range accuracies of less than ten feet. Figure 1 in the supplement to this report (SID62-834-2) indicates the accuracy with which the ephemeris may be determined by the use of radar tracking data of the FPS 16 quality. Prediction of future target position is a function of ephemeris error growth. The accuracy with which the ephemeris data is obtained is a function of basic determination methods and elapsed time since data collection. If sufficient time is spent determining an ephemeris with small error, the error growth rate may be expected to be correspondingly small. Figure 1 in the supplement shows that, after 4 hours of tracking time, the ephemeris error is 1/2 nautical mile. Assuming linear error growth, ephemeris error would be 1 nautical mile 1 hour later. This indicates that midcourse corrections should be delayed as long as possible to determine accurate ephemeris and subsequently to reduce to reasonable values the relative position errors in the vicinity of expected intercept. It is logical that the effective error at the target, which results from errors introduced by the midcourse correction, will decrease as correction initiation time is delayed. However, the magnitude of the required maneuver will increase with attendant propellant requirement increase as time from injection increases. Since the magnitude of the required correction effects a proportional error at intercept, it is obvious that a trade-off is indicated. Consideration of these factors should result in application of the midcourse correction to optimize maneuver magnitude and miss distance. It seems likely that a broad time interval, rather than a

Table 4-1. Radar Data

FPS-16	FPQ-6
Position accuracy	Position accuracy
Azimuth angle = 0.1 mil rms*	Azimuth angle = 0.1 mil
Elevation angle = 0.1 mil rms	Elevation angle = 0.1 mil
Range = 5.0 yd rms	Range = 5.0 yd
Rate accuracy (20 sec smoothing)	Rate accuracy (2 sec smoothing)
Azimuth = 0.0025 mil/sec	Radial rate = 3 ft/sec
Elevation = 0.0025 mil/sec	Tangential rate = 30 ft/sec
Range = 0.3 ft/sec	Rate accuracy (30 sec smoothing)
Rate accuracy (with 2 sec smoothing)	Radial rate = 0.1 ft/sec
Range = 5 yd	Tangential rate = 1.0 ft/sec
Range rate = 3 ft/sec	
Rate accuracy (with 30 sec smoothing)	
Range = 0.5 ft	
Range Rate = 0.1 ft/sec	
*Root mean square	

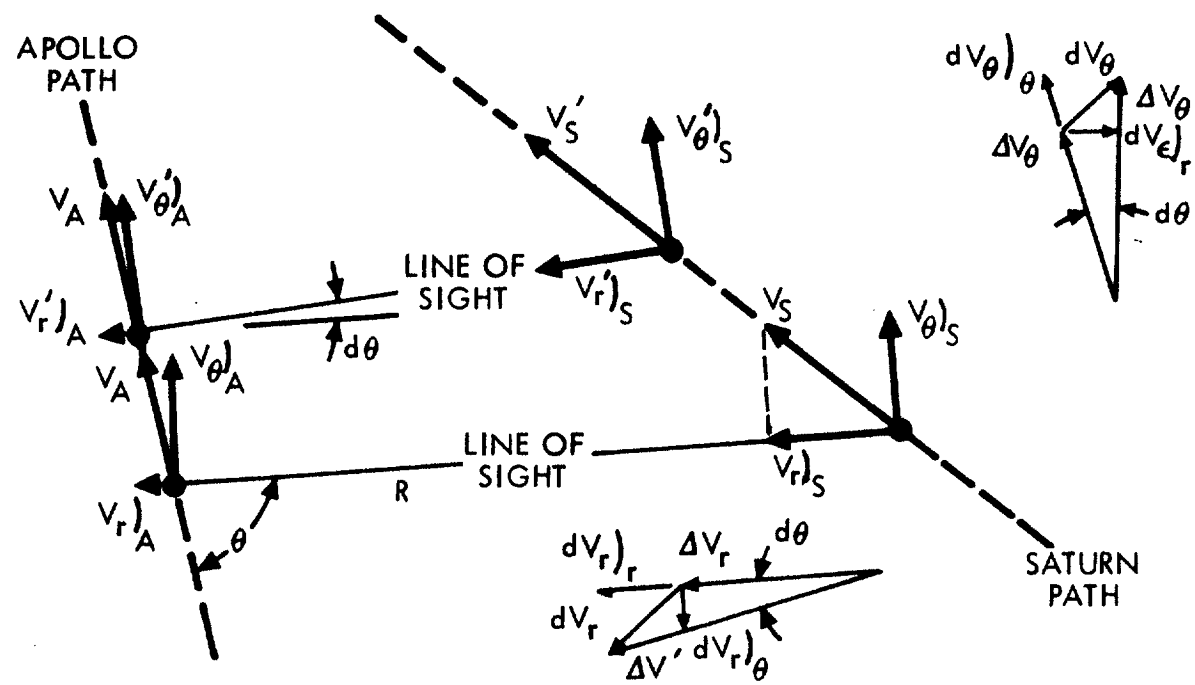
specific time, exists wherein application of a midcourse correction will yield optimum results in the target vicinity.

Terminal Maneuver

Requirements for midcourse guidance stipulate that the intercept vehicle arrive in the target vicinity within the Apollo vehicle-sensor acquisition range. Approach geometry and intercept vehicle parameters must allow completion of rendezvous. S&ID studies have indicated that it is desirable, from a performance standpoint, that the Apollo vehicle approach and establish the proper orbit and lead the S-IVB vehicle with a negative range rate. Thus, the S-IVB will overtake the Apollo, and only positive increments of velocity will be required of the intercept vehicle. Rendezvous requires attainment of zero relative range rate when range becomes zero. Numerous guidance schemes employing relative range, range rate, line of sight, line-of-sight rate, etc., are available; the choice depends on consideration of the possible variety in arrival conditions, propellant requirements, time, and realistic mechanization approaches.

It is recommended that a proportional guidance scheme be employed in the terminal maneuver. This type of guidance is readily adaptable to rendezvous problems, and it ensures that the two vehicles will approach a collision course. Proportional guidance may be accomplished by application of corrective maneuvers proportional to the line-of-sight rate; and, since the relative velocity must be zero when the two vehicles meet, corrections to reduce closing velocity must also be applied. The correction maneuver, then, consists of nulling both line-of-sight rate and relative velocity. The resultant force is composed of one component along the line of sight and another normal to the line of sight.

When the target has been acquired and lock-on has been established by the intercept vehicle, terminal maneuvers may be initiated. To decrease the time required for rendezvous and to enhance the efficiency of the operation, a correction is first applied to null the line-of-sight rate. Essentially, the majority of position errors resulting from incorrect relative velocity normal to the line-of-sight are removed, and a collision course is thus established and maintained. The following sketch presents basic concept:



$V_A = V'_A$

$V_S = V'_S$

$\Delta V_r = V_r)_A - V_r)_S$

$\Delta V'_r = V'_r)_A - V'_r)_S$

$\Delta V_\theta = V_\theta)_A - V_\theta)_S$

$\Delta V'_\theta = V'_\theta)_A - V'_\theta)_S$

$$dV_r = dV_r)_r \mp dV_r)_\theta = \frac{dr}{dt} + \Delta V_r d_\theta$$

$$dV_\theta = dV_\theta)_r \mp dV_\theta)_\theta = \Delta V_\theta d_\theta + R\frac{d\theta}{dt}$$

and

$$A_r = \frac{dV_r)_r}{dt} - \frac{dV_\theta)_r}{dt} = \frac{d^2r}{dt^2} - \Delta V_\theta \frac{d\theta}{dt} = \frac{d^2r}{dt^2} - R\left(\frac{d\theta}{dt}\right)^2$$

$$A_\theta = \frac{dV_\theta)_\theta}{dt} + \frac{dV_r)_\theta}{dt} = R\frac{d^2\theta}{dt^2} + \frac{dr}{dt}\frac{d\theta}{dt} + \Delta V_r \frac{d\theta}{dt} = R\frac{d^2\theta}{dt^2} + 2\frac{dr}{dt}\frac{d\theta}{dt}$$

Reduction of line-of-sight rate is accomplished by orienting the thrust vector normal to the line-of-sight and by applying a velocity increment equal or proportional to the range times the line-of-sight rate. If engine gimbal limits do not enable thrust vector orientation, the vehicle must be reoriented normal to the line of sight, so that the velocity increment can be properly applied. The major objective of the initial maneuver is to reduce the line-of-sight rate to a small tolerable value that may be constantly controlled throughout the remainder of the terminal maneuver. Apollo vehicle sensors furnish the range and the line-of-sight rate; the computer calculates the required velocity increment; and thrust is terminated when the computed velocity increment equals the velocity increment sensed by the Apollo inertial instruments. Precautions must be observed to continuously maintain a negative range rate. In practice, the applied velocity increment is modified by a convenient factor consistent with practical mechanization. This maneuver allows a relatively rapid range closure while a collision course is being established, and the time required for terminal maneuvers is significantly decreased.

When the terminal maneuver has progressed to a certain point determined by consideration of relative range and range rate, a series of corrections are initiated to progressively reduce the relative range rate as the range decreases. Corrections are also continually made to maintain a collision course. The objective of this maneuver phase is to reduce relative range and range rate to roughly 100 to 200 ft and 3 to 5 ft per sec. Once these conditions have been established, the final or vernier maneuver may be accomplished wherein the two vehicles are brought in to proximity (essentially zero range and zero range rate), and the docking operation is performed.

The terminal maneuver phases are outlined in Table 4-2.

Table 4-2. Terminal Maneuver

Phase I	Phase II	Phase III	Phase IV
L-O-S Rate <u>Reduction</u> Initiation Range 50 to 100 Naut Mi	Range Rate <u>Reduction</u> Initiation Range 2 to 5 Naut Mi	<u>Vernier</u> Initiation at Range 100 to 200 feet Range Rate 3 to 5 ft/sec	<u>Docking</u>

VEHICLE CONFIGURATION

Apollo Vehicle

Refer to SID 62-834-2, supplement to this document.

S-IVB Vehicle

The S-IVB will be boosted into an approximate four-hundred mile earth orbit and will subsequently boost the Apollo vehicle into a lunar trajectory. Present indications are that the S-IVB stage will be 58 feet high and 260 inches in diameter and have a propellant weight of 230,000 pounds. Very little is known about the guidance and control systems; consequently, a representative system based upon over-all requirements will be considered. From the standpoint of a rendezvous problem involving the S-IVB and Apollo vehicles, it is desirable that hardware requirements solely applicable to the rendezvous problem be held to a minimum. That is, on-board equipment used for the normal vehicle mission spectrum should be equally applicable to the rendezvous problem with minor additions if necessary.

The multistage Saturn vehicle guidance and control system must be considered with reference to the many possible missions it may perform, including orbit injection, escape trajectories, and reentry flights. The guidance system must also be flexible enough to accept last minute changes in mission or flight profile.

As in other similar systems, the guidance equipment must successfully define a flight trajectory that will meet mission objectives in a near optimum manner (minimum propulsion expenditure). Accomplishment of this goal requires position determination and velocity vectors in some fixed reference system and subsequent computation of required correction terms. The control system, directed by the guidance system, provides and maintains thrust vector and vehicle attitude orientation.

It is likely that an all-inertial guidance and control system will be used in the Saturn vehicle with possible provisions for insertion of pertinent data via radio command if necessary. This choice seems logical because an inertial system can provide the necessary ascent accuracy; has flexibility to readily accommodate the anticipated mission variety; and, being completely self-contained, requires no ground station support. The use of radio command techniques only would be ruled out because launch condition flexibility would be seriously curtailed (a high number of stations would be needed for continuous rf guidance if various launch directions are considered). Provisions for guidance aided by limited radio command are desirable, however, because portions of any particular flight may be accessible to radio command techniques. Discrete corrections in vernier guidance phases necessary to compensate for small inertial sensor errors may then be readily introduced by means of an r-f link with the guidance system. Information derived from radar, radio-doppler, etc., can be evaluated and processed by a ground computer; necessary corrections can be determined; and r-f commands can be used to effect the desired flight profile change. Figure 3 in the supplement to this report shows a basic conceptual block diagram of the assumed Saturn vehicle guidance and control system based on these considerations.

The platform is a four-gimbal, gyro-stabilized unit that provides the necessary freedom for any kind of flight maneuver. Three mutually perpendicular accelerometers furnish incremental velocity information to the digital-computer through the adapter unit. The three gyros have a single degree of freedom; the accelerometers are pendulous integrating units. A chain of platform resolvers furnishes the proper attitude readout for pitch, yaw, and roll. A body-fixed accelerometer is used to measure longitudinal direction acceleration (an indication of instantaneous thrust to mass ratio); the adapter unit performs the necessary processing of computer input and output signals, including digital to analog, analog to digital, switching logic, torquing signal conversion r-f commands, computer-controlled discretes, etc. The use of an r-f command link (digital command) requires an onboard receiver, a digital decoder, necessary computer storage devices, and attendant telemetry equipment for command execution verifications. The control computer uses analog techniques in combining attitude angle commands, rate gyro information, and position feedback to generate the appropriate servoactuation signals.

Contemplated mechanization envisions the use of adaptive guidance techniques in which the system attempts to determine an optimum flight path based on current conditions and desired terminal conditions. No attempt is made to adhere to a standard, predetermined trajectory, but rather to select an optimum trajectory, for the remainder of the flight, based upon current vehicle coordinates. The system is adaptable to a number of mechanization schemes and provides the desirable flexibility for the contemplated variety of Saturn missions.

Major areas of concern for the rendezvous problem include placement of the S-IVB in the proper orbit and S-IVB-Apollo vehicle integration when rendezvous has been completed. Additional areas worthy of consideration involve possible attitude orientation of the S-IVB while in orbit and the addition of S-IVB-borne equipment to aid acquisition by the Apollo vehicle.

Existing methods of orbital injection include programed attitude, radio command, and inertial systems. Precision orbital determination does not appear to be of major importance, because after establishment of the orbit, radar tracking data will be used to determine S-IVB vehicle motion prior to launch of the intercept vehicle. Orbital definition does, however, require a sufficient amount of sophistication to rule out the relatively crude open-loop guidance associated with existing programed attitude methods.

Current best estimates of Saturn guidance and control (Figure 3) specify an inertial system capable of completely self-contained closed-loop operation or, conditions permitting, operation as an inertially aided system. The inertial platform maintains a stable coordinate system; acceleration is measured in this coördinate system and provides continuous indication of the forces affecting the flight profile. Such measurements provide the raw information necessary for generation of guidance errors. A self-contained system is not subject to the line-of-sight restrictions associated with ground control guidance systems; ultimately, performance depends upon accuracy and reliability of the airborne equipment. Performance, of course, degrades as a function of time because no alignment updating is possible for this system subsequent to launch. Including provisions for utilization of r-f techniques results in an inertially aided system in which it is possible to monitor the vehicle flight profile and to introduce corrections for inertial instrument errors. Table 4-3 lists typical weights and power requirements associated with a system of the assumed configuration. The Saturn guidance and control system is capable of performing the orbital injection mission within accuracy requirements.

Table 4-3. Weight and Power Requirements

System	Weight (Pounds)	Power Load (Watts)
Inertial platform	20 to 30	50 to 75
Platform electronics	50 to 60	180 to 190
Adapter unit	25 to 30	20 to 30
Computer	20 to 25	60 to 70
Rate gyros and accelerometer	3 to 5	10
Control system electronics	20 to 30	10 to 20
	Total 138 to 180	Total 330 to 395

Assuming adequate line-of-sight conditions, radio guidance techniques are also capable of satisfactory S-IVB orbit injection. Radio guidance is accomplished by measuring vehicle trajectory with ground based radar, by computing deviations from desired trajectory, and by generating appropriate steering commands accepted by a vehicle-borne receiver and acted upon by the control system. Radio guidance results in significant savings in vehicle-borne systems weight and cost, considerations that may be more important for this particular vehicle mission than for other possible missions. For the present, the Saturn guidance and control will be assumed to be as previously described.

INFRARED AND VISIBLE GUIDANCE

Infrared and visible guidance sensors are examined so that a satisfactory acquisition and line-of-sight angular rate and angle sensor for rendezvous can be selected. The selected sensor system can be used either as the primary sensor or as a backup to a rendezvous radar.

Target Characteristics

The S-IVB skin is internally insulated aluminum painted with white titanium oxide. The incident solar radiation and conduction from the liquid oxygen and liquid hydrogen fuels establish the skin temperatures. The skin temperature minimum will be greater than 185 K and will occur on the dark side of the earth. The painted skin emmissivity is 0.5 and the reflectivity is 0.7. The Saturn is 91 feet high and 220 inches in diameter and presents a minimum target cross section of 24.5 m^2.

Background Characteristics

Sun

Solar radiation is effectively a constant above the earth's atmosphere. The total solar radiation is 0.14 watts per cm^2 with 0.042 watts per cm^2 falling in the visible region and only 4.2×10^{-4} watts per cm^2, falling in the region below 4 microns. The sun subtends an angle of 0.5 degrees in the vicinity of the earth. Sun-protected circuitry must be provided to prevent detector damage.

Earth

The earth can be considered to be a grey body at 273 K with an emmissivity of 0.9. The polar regions will act as comparatively high intensity radiation reflectors. No false lock-on problems should be encountered with the earth during acquisition and the early stages of rendezvous because the Apollo sensors will always be oriented upward to the S-IVB and, hence, away from the earth. As the S-IVB/Apollo range decreases, the S-IVB signal becomes greater. This factor, coupled with discrimination techniques subsequently described, will preclude false earth lock-on during the later stages of rendezvous when the Apollo can be in any position relative to the S-IVB.

Other Celestial Sources

Few stars and planets act as discernible point sources of radiation. The moon and Venus are the brightest sources. Table 4-4 presents the celestial body discrimination problem for selected spectral regions and detection thresholds.

Table 4-4. Celestial Sources Detectable for a Minimum Threshold Signal

Threshold (w/cm^2)	2 to 3μ Detector	3 to 5.5μ Detector	2 to 5.5μ Detector
10^{-13}	13 stars, Sun, Moon, Mercury, Venus, Mars, and Jupiter	8 stars, Sun, Moon, Mercury, Venus, Mars, and Jupiter	13 stars, Sun Moon, Mercury, Venus, Mars, and Jupiter
5×10^{-13}	3 stars, Sun, Moon, Venus, Mars, and Jupiter	2 stars, Sun, Moon, and Venus	5 stars, Sun, Moon, Venus, Mars, Jupiter, and Mercury
10^{-12}	1 star, Sun, Moon, and Venus	No stars, Sun, Moon, and Venus	2 stars, Sun, Moon, and Venus

The moon, when full, subtends an angle of 0.5 degree and emits 3×10^{-7} watts per cm^2 at 70,000 feet.

There are 15,000 known variable stars and 6,300 of these vary rapidly. Starlight, when considered as a single extended source rather than as individual sources, gives a visible flux of 10^{-10} watts per cm^{-2} per $sterad^{-1}$ with spectra similar to that of the sun. The magnitude of all night sky effects except aurora and reflected earthshine is equivalent to 300 tenth magnitude stars per square degree.

The aurora consists of sporadic large, predominantly green, luminous regions beginning at altitudes of 50 miles and extending usually to altitudes of 60 to 100 miles, but rarely as high as 700 miles. The maximum frequency of aurora occurs at a geomagnetic latitude of 67 degrees. The intensity of aurora, as observed from ground level, ranges from 10^{-14} to as much as 10^{-8} watts per cm^{-2} per $sterad^{-1}$. The higher intensities occur during periods of considerable solar storm activity, when it is not feasible to launch astronauts because of the high radiation levels.

There are 7.5×10^{10} meteors of thirteenth magnitude or brighter each day. These meteors are primarily confined to a narrow region of the sky and possess very rapid motion compared to an orbiting space vehicle.

SEEKER AND SPECTRAL REGION SELECTION

The selection of a rendezvous seeker should be based on consideration of the following criteria:

A 50- to 100-nautical-mile Acquisition Range

This range minimizes Apollo rendezvous propellant requirements by permitting removal of relative rates over long time periods.

Range and Range Rate Outputs

The requirement for zero relative velocity occurring at completion of rendezvous dictates a ranging capability.

Angle and Angular Rate Outputs (With Respect to Inertial Space)

The guidance mechanization will require an accurate angular line-of-sight rate. Line-of-sight angle may not be required continuously, but it is needed for initial search pointing. The accuracy of the angle readout need not be high.

Large Gimbal Freedom in Two Axes

Large gimbal freedom is necessary to cover all possible S-IVB and Apollo trajectories and provide the required search volume. Two-axis freedom is desirable because maneuver times can be decreased if the Apollo need not be continuously rolled into the plane formed by the line-of-sight rate and the S-IVB velocity vector. (Rolling into the line-of-sight plane is a requirement of a one-axis seeker.)

Minimum Power, Size, and Weight Requirements

Seeker power, size, and weight all ultimately affect the Apollo total weight.

The Generation of Minimum Seeker Noise and Target Tracking Noise

Noisy tracking data resulting from S-IVB motion, tracking limitations, or inherent seeker noise tends to degrade rate information. For example, if the position noise content is high and varying randomly, the noise will produce false rate data that will have to be filtered out; guidance lags are thereby introduced. Then if the second order time derivatives of range and line-of-sight angles are used in mechanizations, the noise problem becomes extremely acute.

Proven Operation or High Probability of Success With Further Development

Reliability is important in flight to increase the probability of mission success. It is important on the ground so that spares and field maintenance are kept to a minimum.

Possession of a Short Blind Range

Short blind ranges permit the use of automatic aides almost to the point of docking and free the pilot for other jobs.

Inertial stabilization of the sensor will be necessary during the initial portion of rendezvous to permit target tracking at the time that thrusting is being performed with a single gimbaled engine. Inertial stabilization of the sensor is also desirable so that Apollo body motions are decoupled from target motions and so that sensor servo-loop stability problems are simplified. At short ranges, low line-of-sight rates exist, and essentially unlimited thrust vector positioning is available because of the use of multiple vernier engines. Short-range operation using a body-bound sensor appears feasible.

Sensor domes should not be a problem because the system can be exposed to the vacuum environment after boost.

It is desirable to avoid detector cooling, if possible, to avoid difficulties in handling liquid nitrogen and helium.

Refer to Table 4-5 for seeker comparisons.

Table 4-5. Rendezvous Sensor Comparison

Characteristic	Visible	Far IR	Active Radar (Uncooperative)	Radioisostope Source and Scintillation Counter
Size (Cu ft)	0.1	0.7	1.5	0.1
Power (watts)	55	60	1100	5
Weight (pounds)	6 to 10	54	120	11
Outputs	Line-of-sight angular rate and angle with respect to inertial space		Line-of-sight rate, angle, range, and range rate	Range and range rate
Acquisition range (nautical miles)	127	46	67	25
Background discrimination problems	Some stars, sun, moon, planets are false targets		Not applicable	Cosmic rays and other background space radiation

Table 4-5. Rendezvous Sensor Comparison (Cont.)

Characteristic	Visible	Far IR	Active Radar (Uncooperative)	Radioisostope Source and Scintillation Counter
Inertial Stabilization	Either use spinning optics, slave to inertial platform, or use gimbal mounted rate gyro in stabilization loops.		Slave to inertial platform-large diameter collector makes spinning optics stabilization impractical; or use gimbal-mounted rate gyro in stabilization loop.	Not applicable
Conclusions	No earth shadow operation; discrimination difficult	Cooled detector necessary; requires exotic materials, difficult optics design (system described has 5 in. optics).	Large power required; system quite heavy and bulky	Inaccurate data during acceleration modes; requires Apollo-mounted beacon

ACTIVE VERSUS PASSIVE SYSTEMS

Active Visible and Infrared

Present active visible and infrared sources have only short range capability (in the order of a few thousand feet to a few miles) because of inefficient energy conversion, which results in high power requirements. The sources must be pulsed to obtain range rate information because the radiation is incoherent. At short ranges, the pulsed ultraviolet, infrared, and visible sources have power, weight, and size advantages over radar. The short wave lengths provide theoretically higher accuracies than microwave techniques.

The laser shows promise of development into an efficient source of very narrow beam coherent radiation suitable for space missions. As such, it could prove to be a real competitor to radar (because of its size, weight, and power advantages), even for long-range acquisition. Laser state of the art does not permit further consideration at this time.

Passive Visible and Infrared

Passive systems have weight, size, power, and range advantages over active systems. Passive systems cannot provide accurate range information, as the following discussion will point out. Three methods of passively generating range information are rangefinders, measurement of focal plane shift, and measurement of the target radiation intensity change. Feasibility of passive infrared rangefinders was studied by S&ID in connection with the Saint proposal. It was determined that by using two high-resolution (1000 line) thermal imaging tubes with a base separation of 5 feet in a zero-vibration environment, the ranging accuracy might be compatible with some rendezvous mechanizations out to the initial detection range of the tubes. Currently, this initial detection range is about 13 nautical miles with reasonably sized optics. The possible range error at 13 nautical miles is ±47 percent rapidly decreasing to ±1 percent at 1000 feet. The ability of such a design to actually obtain these accuracies in practice has not been proven. The measurement of focal plane shift is actually another method of range finding and is subject to the same magnitude of inaccuracy.

Intensity ranging is not reliable in this situation because seeker intensity is varying not only as a function of Apollo-S-IVB range, but also as a function of S IVB sun aspect angle. The use of controlled-intensity beacons on the S-IVB could possibly make this method of ranging feasible.

Imaging Tubes, Mosaics, and Point Detectors

Imaging tubes have features that warrant their consideration for use as homing seekers. (Imaging tubes are vidicons or image orthicons with photosensitive surfaces matched to the appropriate spectral region.)

Advantages

The advantages of imaging tubes are the following:

1. Search-scanning and tracking can be performed electronically.
2. Track-while-scan and multiple target viewing are feasible.
3. Airborne discrimination can be enhanced by moving target indication (using long and short persistence phosphors), two-color operation, or target size and shape indication.
4. The shape indication can be used to obtain a crude measure of range.
5. Sensitivity is adequate for sensing the expected level of S-IVB radiation.
6. Blind range is only a few feet.

Disadvantages

The disadvantages of imaging tubes are the following:

1. The outputs of an imaging tube are S-IVB azimuth and elevation angles. Rate information is electrically differentiated position data using a Δt equal to the frame time of the electronic scan plus or minus the target rate (depending on sign). This method of obtaining rate information is both sluggish and noisy.
2. The seeker output contains Apollo body rates in addition to S-IVB inertial line-of-sight rates if the seeker is mounted to the body. (Elimination of gimbals is one of the chief advantages. Refer to the discussion on the advantages of seeker inertial stabilization in a following section.)
3. For a reasonable seeker search field of view (field-of-view size affects background discrimination), it would probably still be necessary to provide a gimbal torquer system for initial seeker

pointing, thus negating some of the advantages of an electronic scanning system.

Comments on mosaic detectors are similar to those applying on the imaging tube. But here an additional disadvantage exists in that the resolution of a mosaic is less than that of an imaging tube, and the electronics are more complex because of the need of many separate amplifiers.

Point detectors require servo-loops to maintain coincidence of the tracker axis and interceptor-to-target line-of-sight vector. This complexity permits the use of small track fields of view. This results in a high signal to noise ratio. The tracking servo-loops can also be used to initially point the seeker.

DETAIL SYSTEM SELECTION

The choice of spectral region and detector go hand in hand, since a spectral region that is advantageous from most standpoints may present detector problems that override the advantages.

Infrared

The chief advantage of infrared techniques is that they can be used to passively acquire the target regardless of sun illumination on the target.

Derivation of the Infrared Range Equation

An expression can be written for acquisition range in terms of many of the design parameters of concern in a study such as this. The resulting equation is lengthy, even though some secondary effects are not considered; the equation, despite its complexity, readily points up the interdependence of parameters.

The steradiancy of the target is obtained using the Steffan-Boltzmann Law:

$$J_2 = \frac{\sigma \epsilon \eta T^4 A_T}{\pi} \text{(watts/steradian)}$$

where

σ = Steffan-Boltzmann constant

ϵ = emissivity of target

η = percentage of total target radiation falling in selected spectral region.

J_2 = steradiancy in spectral region of interest

A_T = cross-sectional area of target (target assumed to act as Lambertian diffuser)

Now, the power density required for detection is

$$P_D = NEP \sqrt{BW} \left[S/N \right] (\text{watts/cm}^2) \text{ or } P = \frac{\sqrt{A_d BW} \quad [S/N]}{D^*} \text{ watts}$$

where

P_D = power density

P = power

NEP_D = noise equivalent power density (watts/cm^2 cps 1/2)

BW = bandwidth

S/N = target signal-to-system-noise ratio

A_d = detector area

D^* = detectivity

or, alternately,

$$P = J\Omega_c K_c K_a K_o = J \frac{A_c}{R^2} K = \frac{J \pi d^2 K}{4R^2}$$

where

Ω_c = solid angle subtended by collector at the target range

R = target range

A_c = area of collector

d = diameter of collector

K_c = chopping efficiency

K_a = atmospheric transmission efficiency = 1 in space

K_o = optical transmission efficiency

Equating the expression for power and substituting for J

$$\frac{\sqrt{A_d BW}\ (S/N)}{D*} = \frac{\sigma\epsilon\eta T^4 d^2 K A_T}{4R^2}$$

Then

$$R = \left(\frac{T^2 d}{2}\right) \frac{\left[\sigma\epsilon\eta K D* A_T\right]^{1/2}}{\left[A_d BW\right]^{1/4} (S/N)^{1/2}}$$

Discussion of Equation

Any infrared range equation can give only an approximation of the actual target range. This is because of the large number of assumptions that must be made about the target and the detector system itself. The derived equation ignores background signal in relations to target signal, tacitly assuming that the spacial and spectral filtering and limited field-of-view can keep the background signal at a low level. Approximations of integral quantities, such as atmospheric transmission, are made to avoid iterative solutions.

No detailed optimization of the infrared system will be made at this time, but an estimate of infrared acquisition range will be made to compare the various sensors. The nominal calculation used the following parameter values.

Steffan-Boltzmann constant - $5.6686 \times 10^{-12} \left(\frac{\text{watts}}{\text{cm}^2 (\text{degK})^4}\right)$

Target data (Target Estimates are Conservative):

T	Target temperature = 183 K min
η	Spectral efficiency = 0.58
	(8-25μ spectral region, 183 K target)
ϵ	Target emissivity = 0.5
A_T	Target cross-sectional area = 24.5m^2 min
d_T	Target diameter = 220 in.

Seeker data:

d	Diameter of collecting optics = 12 in. max
K	Transmission factor = 0.25
	(optical efficiency, 0.5, x chopping efficiency, 0.5, x
	Atmospheric transmission, [1])
D*	Detector detectivity 1 x 10^{10} cm cps 1/2 watt
	(copper doped germanium, 15°K)
A_d	Detector area = 0.01 cm^2 min
BW	Amplifier bandwidth = 100 cps
S/N	Receiver required signal-to-noise ratio = 10

Substituting these values in the infrared range equation gives an acquisition range of 109 nautical miles. This range is obtained with the large collecting optics diameter of 12 inches resulting in a great amount of expense and weight. Furthermore, the detector must be cooled to 15°K, this necessitates liquid-helium cooling with its attendant complexities and operating time restriction. An S/N of 10, while not generous, should result in adequate background discrimination. (See Table 4-4.) The calculation is conservative since no earth albedo was included in the steradiancy calculation and the minimum target crossection and skin temperature were used. It should be reiterated that this system is completely passive and is capable of operating regardless of S-IVB location with respect to the earth's shadow.

Visible

Use of visible radiation detectors requires that the S-IVB be in sunlight during acquisition and rendezvous if passive devices are to be employed. The S-IVB can be in darkness up to 40 percent of its orbit period during particularly unfavorable orbits. The advantages of using the visible spectrum are the following: (1) the reflected sunlight radiation level is higher than the S-IVB self-emission radiation level; (2) visible spectrum detectors are on the average more sensitive than infrared detectors; and (3) visible optical designs are easier to make and are more efficient because optical corrections need not be made over as wide a wavelength region as is required for infrared optics.

Visible Range Equation

Only minor changes are required in the infrared range equation to make it suitable for use in the visible spectrum. These changes will be indicated here. Terms that correspond with their infrared counterparts are not redefined here. In particular, the equation is suitable for use with photomultiplier tubes. The quantity of solar radiation diffusely reflected from the target is given by

$$R_{\lambda} = \frac{P_S \rho A_T \eta}{\pi} \quad \text{(watts/steradian)}$$

where

P_S = solar constant (watts/cm^2)

ρ = reflectivity of target skin

R_{λ} = reflected solar power in spectral region corresponding to η - analagous to J_{λ}

P = NEI (S/N) (watts)

NEI = noise equivalent input (watts)

= NEI (lumens) $\times \frac{1}{\eta_{\epsilon}'}$

η_{ϵ}' = temperature dependent conversion from photometric units to watts.

Alternately,

$$P = \frac{R_\lambda \pi d^2 K}{4R^2}$$

Equating the expressions for power and substituting for R ,

$$NEI\ (S/N) = \frac{P_S \rho A_T \eta d^2 K}{4R^2}$$

Then

$$R = \frac{d}{2}\left[\frac{P_S \rho A_T \eta K}{NEI\ (S/N)}\right]^{1/2}$$

Target data:

ρ	Target reflectivity = 0.7
P_S	Solar constant = 0.14 watts/cm^2
η	Spectral efficiency = 0.22
A_T	Minimum target area illuminated by sun = 1m^2 (quite conservative)

Detector data:

Type = 1P21 photomultiplier

Surface = S4

NEI = 5×10^{-13} lumen

N_e^1 = 83 lumens/watt

S/N = 100 (minimizes discrimination problems)

Visible detection range:

$$\frac{R}{d} = \frac{1}{2}\left[\frac{1400(0.7)(1)(0.22)(0.25)83}{5 \times 10^{-13}(100)}\right]^{1/2} = 4.73 \times 10^{6}$$

or

$$R\,(\text{nm}) = 63.6\ d\ (\text{in.})$$

so that 2-inch optics provide a conservative detection range of 127 nautical miles.

BASIC CONSIDERATION OF SITUATION EXISTING AT SENSOR ACQUISITION

Assume a two-dimensional problem neglecting earth gravity effects and system response errors. (Refer to vector diagram under terminal maneuver.) The resulting equations of motion are the following:

Relative accel along los $= \ddot{R} - R\dot{\theta}^2$

Relative accel normal los $= R\ddot{\theta} + 2\dot{R}\dot{\theta}$

Relative velocity along los $= \dot{R}$

Relative velocity normal los $= R\dot{\theta}$

The intent of the initial maneuver is to cause $V_\theta)_A$ to equal $V_\theta)_S$; if this could be caused to occur by adding a velocity increment to $V_\theta)_A$, the line-of-sight rate could be reduced to zero.

At any instant, the required increment of velocity is seen to be equal to $R\dot{\theta}$.

In practice, the line-of-sight rate is reduced and maintained at an acceptably low level by the use of range and LOS rate data until the range reaches a preselected suitable value at which time additional maneuvers designed to reduce range rate are initiated.

V. ELECTRONIC SYSTEMS

TRACKING EARTH-ORBITING VEHICLES FOR SPACE RENDEZVOUS

The probable necessity of orbiting the S-IVB and the Apollo manned vehicles separately and then rendezvousing in space to join the two vehicles prior to a lunar trajectory launch will necessitate that certain tracking requirements be met. The actual parking orbit of the initial vehicle will have to be ascertained prior to the launch of the second vehicle; and, since it is an unmanned vehicle, the S-IVB will undoubtedly be under gound control until it is mated with the Apollo vehicle. When the two vehicles have been joined and the S-IVB has been ignited, additional tracking will be required to ascertain that the correct lunar trajectory has been established.

Two orbital altitudes are planned for the rendezvous. A 90-nautical-mile parking orbit will be established for one or both of the vehicles. Actual rendezvous will be accomplished at an orbital altitude of approximately 300 nautical miles. (See figure 5-1 at the end of this section.)

Mercury Tracking Network

Any discussion of world-wide tracking systems for orbiting vehicles automatically begins with the operating Mercury network. This 16-station network was specifically designed for the three-orbit Mercury program; any attempt to use it in a 16-to-18 orbit per day program will necessitate certain trade-offs of information data and tracking requirements. The original goal[1] on Mercury network was to have a period of no longer than ten minutes during which the astronaut could not communicate with the ground. This has not been met in practice, since there is a 13 minute gap in the first orbit, 12 and 15 minute gaps on the second orbit, and two successive 30-minute gaps on the third orbit. Successive orbits will have even larger time gaps, and some orbits will not even pass through the present (1962) coverage area of a Mercury station.[2]

Mercury station locations and their equipment capabilities are listed in Table 5-1.

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1 Satellite Instrumentation Network Facilities. NASA, Goddard Space Flight Center.

2 Preliminary Performance and Interface Specifications, GOSS, Project Apollo. NAA S&ID, SID 62-76 (28 Feb. 1962).

Table 5-1. Mercury Tracking Stations

Number and Location of Site	AN/FPS-16	Verlort	Acquisition Aid	Telemetry-Voice
1. Cape Canaveral	X	-	**	X
2. Bermuda	X	X	X	X
3. Mid-Atlantic Ship	-	-	X	X
4. Grand Canary Island	-	X	X	X
5. Kano, Nigeria	-	-	-	X
6. Zanzibar	-	-	-	X
7. Indian Ocean Ship	-	-	X	X
8. Perth, Australia	-	X	X	X
9. Woomera	X	-	X	X
11. Canton Island	-	-	-	X
12. Kokee, Kauai, Hawaii	X	X	X	X
13. Point Arguello	X	X	X	X
14. Guaymas, Mexico	-	X	X	X
15. White Sands	X	-	X	-
16. Corpus Christi	-	X	X	X
17. Eglin Air Force Base	X	*	X	-

* Eglin has an AN/MPQ-31 instead of Verlort

** Cape Canaveral uses a TLM-18 telemetry tracker as an acquisition aid.

Except at Cape Canaveral, the acquisition aid will be a quad-helix antenna used in a manner similar to a monopulse radar antenna. It will be able to track continuously on any telemetry transmitter or voice signal. Its broad pattern (approximately 20 degrees beam width) can easily acquire the vehicle; telemetry, voice, and radar antennas are slaved to the quad-helix antenna; and when the vehicle is acquired, automatic track is instituted. The TLM-18 antenna used at Cape Canaveral (and Antigua and Ascension on the AMR range) is a 60-foot parabola capable of a 28-db average gain; the quad-helix antenna average gain is 18 db.

The AN/FPS-16 is a C-band precision radar with a 1-megawatt output and a nominal 500-nautical-mile range (1000 at Bermuda) and is used in conjunction with a transponder in the vehicle. Verlort is an S-band radar system with a peak power of 250 kilowatts and a 1000-nautical-mile range, in conjunction with a transponder in the vehicle.

The Mercury network is linked together by land lines, submarine cables, and microwave links. All data are fed to the NASA computation center in Greenbelt, Maryland, where orbital computation is accomplished. Orbital information is also disseminated to all network stations for assisting in tracking operations. Certain stations on the AMR can be used by feeding data through the Cape Canaveral Control Center.

Radar Track Capabilities, Mercury Network

Radar tracking in the Mercury network is done in a beacon tracking mode by two basic radar systems AN/FPS-16 and Verlort. (Instead of Verlort, Eglin uses a modified AN/MPQ-31, which is similar to Verlort in performance and it has a digital output similar to the FPS-16 output.) The two radar systems are used to interrogate a transponder in the vehicle by transmitting a coded pulse train and to track the range and angle of the response. Antennas of both radar systems are slaved to the acquisition aid for initial directional information. Both systems provide continuous azimuth coverage; elevation coverage is -10 to 85 degrees for the FPS-16 and -11 to 90 degrees for the Verlort.

Only 11 of the 16 Mercury stations have radar tracking capability. Six of these stations are in the continental United States or just south of the Mexican border (Guaymas) and, with Bermuda and some AMR stations, provide continuous coverage of approximately 95 degrees longitude. Coverage patterns of the Hawaiian Verlort station and the Point Arguello Verlort station fail to overlap by about three hundred miles. There is a hole of at least seven hundred fifty miles between the coverages of the Bermuda and Grand Canary Island stations. The other two radar stations are located in Australia and have overlapping coverage patterns. Consequently, there is no radar coverage between West Africa and west of Australia and between east of Australia and southwest of Hawaii.

Radar Improvement, Mercury Network

Improvement of the range tracking capabilities of both FPS-16 and Verlort radars has been undertaken, but present progress is not known. Kits are being manufactured and distributed to increase FPS-16 power to 3 megawatts peak and to improve receiver sensitivity. This modification (already accomplished at White Sands) will allow for C-band automatic tracking out to 7000 nautical miles if the vehicle system has a transmitter power of 155 watts and a receiver sensitivity of -65 dbm (both presently available in a single unit).[3] Small also reports that modification of the Verlort system is underway to increase the power to 5 megawatts and the antenna gain to 52 db; this improved system will have tracking capability to 7000 nautical miles if vehicle transmitter power is 125 watts.

The radar improvement program will be beneficial during the early stages of the Apollo lunar trajectory but will not materially increase tracking ranges during the 90-mile parking orbit phase of the program. During the 300-mile rendezvous orbits, tracking ranges will be almost doubled by the improvement program over the present range capabilities. These coverages will be discussed subsequently.

Telemetry Track Capabilities

All Mercury network stations, except White Sands, have telemetry and voice communications capability. The five stations without radar systems have telemetry-voice capability and, except Canton Island, acquisition aids. Therefore, these stations can provide azimuth and elevation angles of the tracked vehicle. These data can be fed to the computer center and compared to predicted orbital data to ascertain that the vehicle position is essentially correct. Any errors can be positively checked when the vehicle enters a radar station coverage area, and corrections can be made as necessary.

Telemetry range depends on transmitted power, atmospheric attenuation, and ground receiving system characteristics. Except at Cape Canaveral, all telemetry stations will be using quad-helix receiving antenna with 18-db gain followed by an antenna tricoupler with 0.4-db loss. This feeds a signal to a vacuum tube amplifier which has a 3.5-db noise figure and

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[3] Small, J. Preliminary Investigation of Ground Tracking and Communication Systems Adaptable for use in Project Apollo. NASA Project Apollo Working Paper 1009 (31 Jan. 1961).

a 26-db gain. From here a 100-foot cable takes the signal into a Nems-Clarke receiver. The difference at Cape Canaveral is the use of a TLM-18 antenna with a 26-db gain and an antenna amplifier with a noise figure of 4.5 db.

Telemetry range of the Apollo and S-IVB vehicles will depend on component selection. The telemetry transmitter for Apollo will have a power output of 10 watts. Three telemetry systems, one voice channel, and one VHF beacon will feed a five-channel multiplexer with 1.5 db insertion loss. Approximately ten feet of coaxial cable will be required, and if RG-122/u is used (small diameter, high attenuation) cable loss will be about 1.2 db. The discone telemetry antenna will be located in the movable nose of the Apollo; a rotating coaxial joint will be required. A special JPL axially rotating joint has a maximum insertion loss of 0.5 db at 960 M c/s; a well-designed swinging-rotation joint for Apollo should stay within 1.0 db insertion loss. Thus, rf transmission system has a loss of 3.7 db.

The telemetry antenna will be a discone. Research of discones[4] indicated average antenna losses of 3-db with 60-degree conical laboratory antennas. Assuming some degradation due to production techniques and "environmentalizing" the discones, a loss figure of 5 db is arbitrarily assigned. This gives total rf systems loss of 8.7 db.

Ground station antennas have a gain of 18 db (except Cape Canaveral). The antenna tricoupler has a loss of 0.4 db and antenna amplifier a noise figure of 3.5 db. If this amplifier is assumed to have the same band width as the Nems-Clarke receiver, the required power input to the amplifier can be found from[5]

$$F = \frac{Pi}{K\ To\ Vf}$$

Where F = noise figure

K = Boltzmans Constant

To = Ambient temperature

Vf = bandwidth

Pi = power input

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[4] Nail, J. J. "Designing Discone Antennas," Electronics (Aug. 1953).

[5] Reference Data for Engineers. Fourth edition.

If the narrowest bandwidth available from the receiver is used, the worse condition requires an input of 4.4×10^{-16} watts. To find the maximum range at which Apollo telemetry systems can transmit signals to standard Mercury network receivers, the following formula can be used:

$$R = \left[\frac{P_t G_t G_r \lambda^2}{(4\pi)^2 P_i} \right]^{1/2}$$

Where P_t = transmitted power

G_t = Gain of transmission system

G_r = Gain of receiving system

λ = Wavelength (250 Mc/s)

Pi = Power input at receiver

R = Range, miles

Substituting the aforementioned quantities into this equation gives a range of 8650 nautical miles for the Apollo telemetry range.

The S-IVB telemetry range should be approximately the same if 10-watt transmitters are used. Slot antennas on the S-IVB should not offer more than a 5-db loss; multiplexer loss will not be more than 1.5 db. Although rotating joints need not be used, longer transmission lines may offset this gain. With a 2-watt transmitter, radiating range should still be on the order of 4300 nautical miles.

Tracking Capabilities, 90-Mile Orbit

Initial parking orbit for the S-IVB and/or Apollo vehicles will be effected at a 90-nautical mile altitude. At this particular altitude, line-of-sight condition is the limiting range factor rather than capabilities of the vehicle or ground equipment. Under the assumption that the ground antenna is at sea level, the line-of-sight range can be found from the formula

$$D = (h^2 + 2\,R\,h)^{1/2}$$

Where D = range in nautical miles

R = average earth radius in nautical miles

h = vehicle height in nautical miles

For this particular orbital altitude, the line-of-sight range is 800 miles.

For multiorbit missions at 90-mile altitude, the Mercury network will not have a track capability on every orbit. Tracking holes are evident in the South Pacific and Indian Ocean areas and, to a lesser extent, in Central Africa. There is no coverage between Eastern Australia and the Canton Island range pattern, and several northwest-to-southeast passes will travel over this area. In the Indian Ocean area, there is a pattern coverage gap between the Indian Ocean Ship and Zanzibar for southwest-to-northeast passes. In Central Africa, there is a lack of coverage between the Zanzibar and Kano areas, affecting the southwest-to-northeast passes. Utilizing approximate information, these holes will cause a loss of track on a vehicle from Hawaii on the eighth orbit to Kano on the tenth orbit; this is approximately one and one-third orbits, about 120 minutes. Two orbits later, as the vehicle passes over Grand Canary on the twelfth orbit, there will be one full orbit of no track until the vehicle again passes over Grand Canary on its thirteenth orbit. On all other orbits, tracking black-out times will be less severe.

A tracking ship stationed off either coast of South America at 30 degrees south latitude would decrease the loss of track time to approximately 90 minutes (just less than one orbit). If this ship were to be stationed at 30 degrees south latitude and 90 degrees west longitude, the second longest blackout period would be cut from 90 minutes to approximately 60 minutes, although it would have no affect on shortening the longest blackout period of one orbit. If NASA were to insist on tracking information at least hourly, two new land stations and six tracking ships would have to be added. These would be located at Durban, South Africa; Noumea, New Caledonia; the mouth of Persian Gulf (25° N 55° E); the mouth of the Gulf of Aden (12° N 52° E); between California and Hawaii (24° N 130° W); a ship off Peru (10° S 83° W); a ship off Chile (28° S 75° W); and a ship off Argentina (32° S 50° W).

Tracking Capabilities, 300-Nautical-Mile Orbit

The S-IVB will rendezvous with the Apollo at a 300-nautical-mile altitude. Using the given formula, the line of sight at this altitude is 1465 miles. In order to track at this range, the radar systems must be improved. Since both Verlort and FPS-16 radar systems are presently being modified and since the Apollo space rendezvous is still 2 to 4 years off, it is a safe assumption that the 1465-mile range is achievable.

With a 300-nautical-mile altitude orbit, the present Mercury network is almost satisfactory. The only major hole in the pattern coverage is between East Australia and Canton Island. This particular hole will cause a near one-orbit blackout on about the eleventh orbit. Otherwise, the vehicle will be passing through the coverage pattern of some tracking station at least every half orbit. Further, there would be an almost continuous track during the first four orbits.

Only one station would have to be added to improve tracking in the 300-nautical-mile orbit. This station would be located at Noumea, New Caledonia. With the addition of this station, the vehicle could be tracked at least every half orbit.

Additional Station Requirements

The inability of the present Mercury network to provide tracking capability at least once in every orbit of a multiorbit mission may not be too serious a problem. If the S-IV is assumed to be boosted into a parking orbit first, the orbit can be easily determined during the first four orbits because it consistently passes through the Mercury stations during these early orbits. Subsequently, spot checks may be made to ascertain if the spacecraft is continuing in its predicted orbit. Once the S-IVB is in orbit, the final countdown on Apollo can begin. Actual firing of the Apollo into orbit will undoubtedly be confined to certain times when the predicted orbit will closely match the S-IVB orbit. This condition will necessitate minimum maneuvering requirements for the two vehicles previous to rendezvous.

Spot-check tracking of the two vehicles should be accomplished at least once during each orbit (preferably more often). To accomplish a once-per-orbit minimum track capability, at least two new stations would be required. One would be a land station located at Noumea, New Caledonia while the other would be a ship stationed approximately 30 degrees south, 90 degrees west. These two stations would allow at least one check per orbit for both the 90- and 300-nautical-mile orbits.

Track capability improvement to once per half-orbit could be achieved by a total of five new stations (including the two recommended). Besides locating the new land station at Noumea and moving the South American ship up to 25 degrees south latitude (and 90 degrees west longitude), a new land station at Durban, South Africa; a ship at the mouth of the Gulf of Aden off Sokotra Island (British); and a ship at the mouth of the Persian Gulf would give such capability. The maximum hole for any orbit would be 180 degrees on the ninth orbit with smaller holes on all other orbits. Basically, the Durban, Zanzibar, Sokotra Island, and Persian Gulf stations present a continuous north-south coverage line around 45 degrees east longitude and would be capable of tracking every pass. Another southwest-northeast line would be presented by the Australian, Noumean, Canton, and Hawaiian stations, with the small holes being backed up by the South American ship station.

Additional tracking improvement could be obtained if stations could be added from the AMR capability. These stations would be at Ascension Island, which already has a FPS-16 radar plus telemetry, and Antigua, which is installing a C-band FPQ-6 in 1962. Both stations could join the network via transmission of data through the Cape Canaveral control center. These two economical additions would add coverage across northern South America, the mid-Atlantic on both sides of the equator, and almost to the lower west coast of Africa.

Implementation of the five additional stations required to ensure effective tracking data at least every half orbit would not be too great an obstacle. Land base stations at Durban and Noumea would have to be built. However, some economy may be realized through proper utilization of existing ships for tracking stations. The present mid-Atlantic ship (28 degrees north, 40 degrees west) could be moved (addition of Antigua could easily replace the ship with little loss of data), preferably to the Persian Gulf location. The Gulf of Aden station could possibly be manned by the Project DAMP[6] ship, USAS American Mariner. This ship already has two 3-megawatt[7] FPQ-4 C-band radars with necessary communications equipment including some data reduction facilities. Telemetry capability is presently unknown, but undoubtedly some equipment is aboard as this

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[6] Project DAMP. Barnes Engineering Company brochure.

[7] Range Communication Instruction Manual. Atlantic Missile Range, MT57-27433.

ship was used in reentry body experiments on the AMR. Improving its telemetry capability should not be too expensive nor should the expense of setting the ship into the Mercury data network be excessive. To man the tracking station off the western coast of South America, a ship similar to the USNS Skidmore Victory could be used very effectively. This Pacific Missile Range ship[8] is fully instrumented with long-range, C-band, precision radar; telemetry equipment; meteorological station; and communications equipment. If the Skidmore Victory should not be available, it is strongly recommended that any ship chosen for this station have a radar tracking capability. This ship will be used on a number of orbits to effect the systems half-orbit tracking capability and should have range tracking capability to provide positive tracking on these orbits. It is also recommended that both land-based stations contain precision radars for similar reasons.

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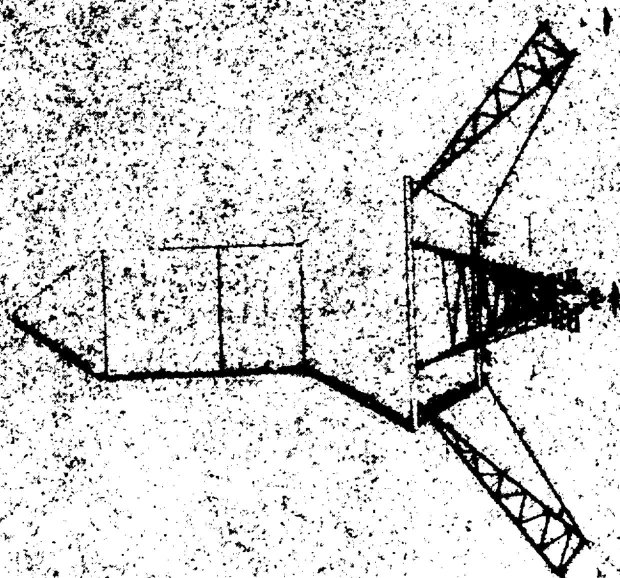
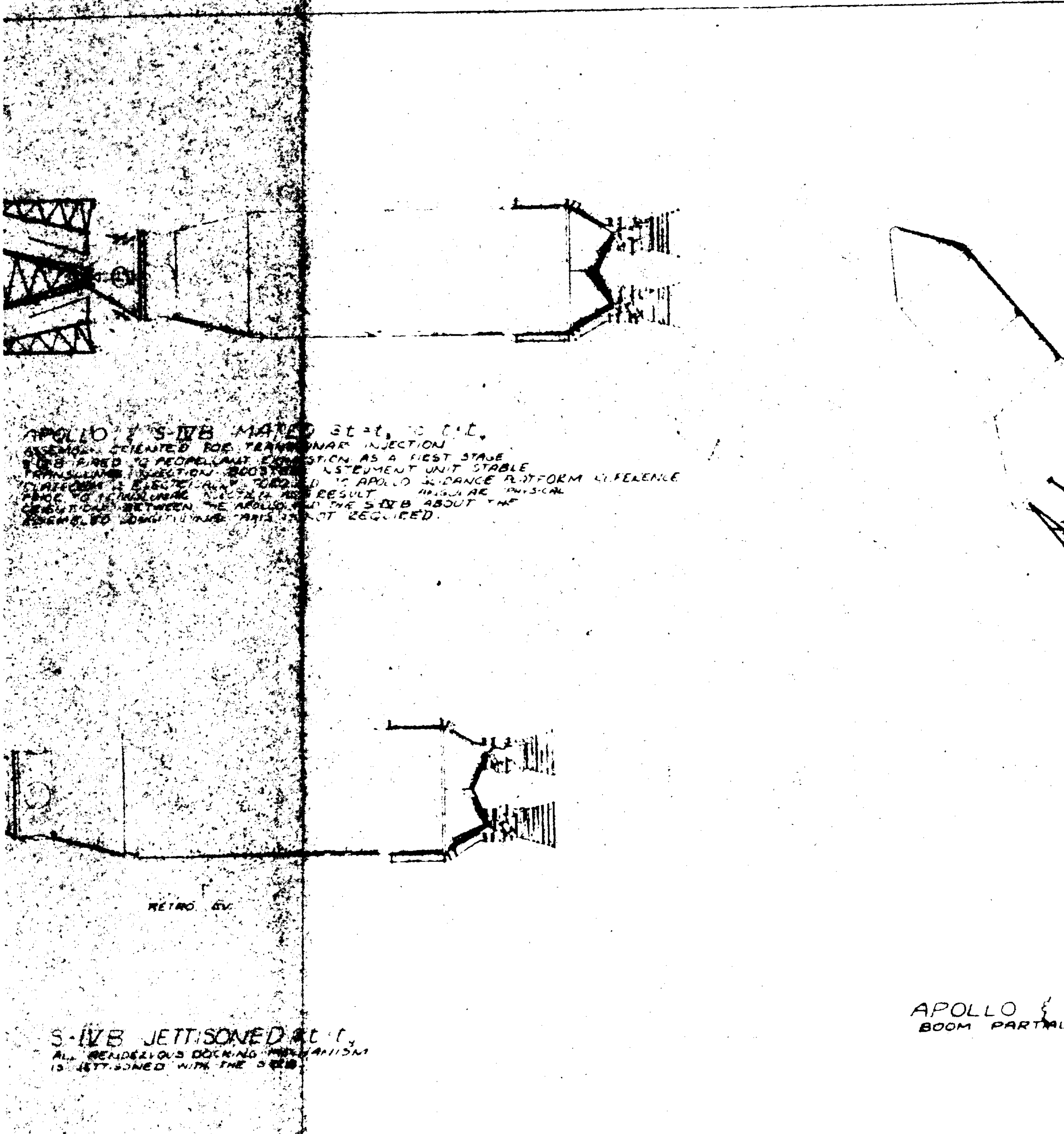
[8] Range Manual. Pacific Missile Range, PMR-MP-60-9.

APOLLO

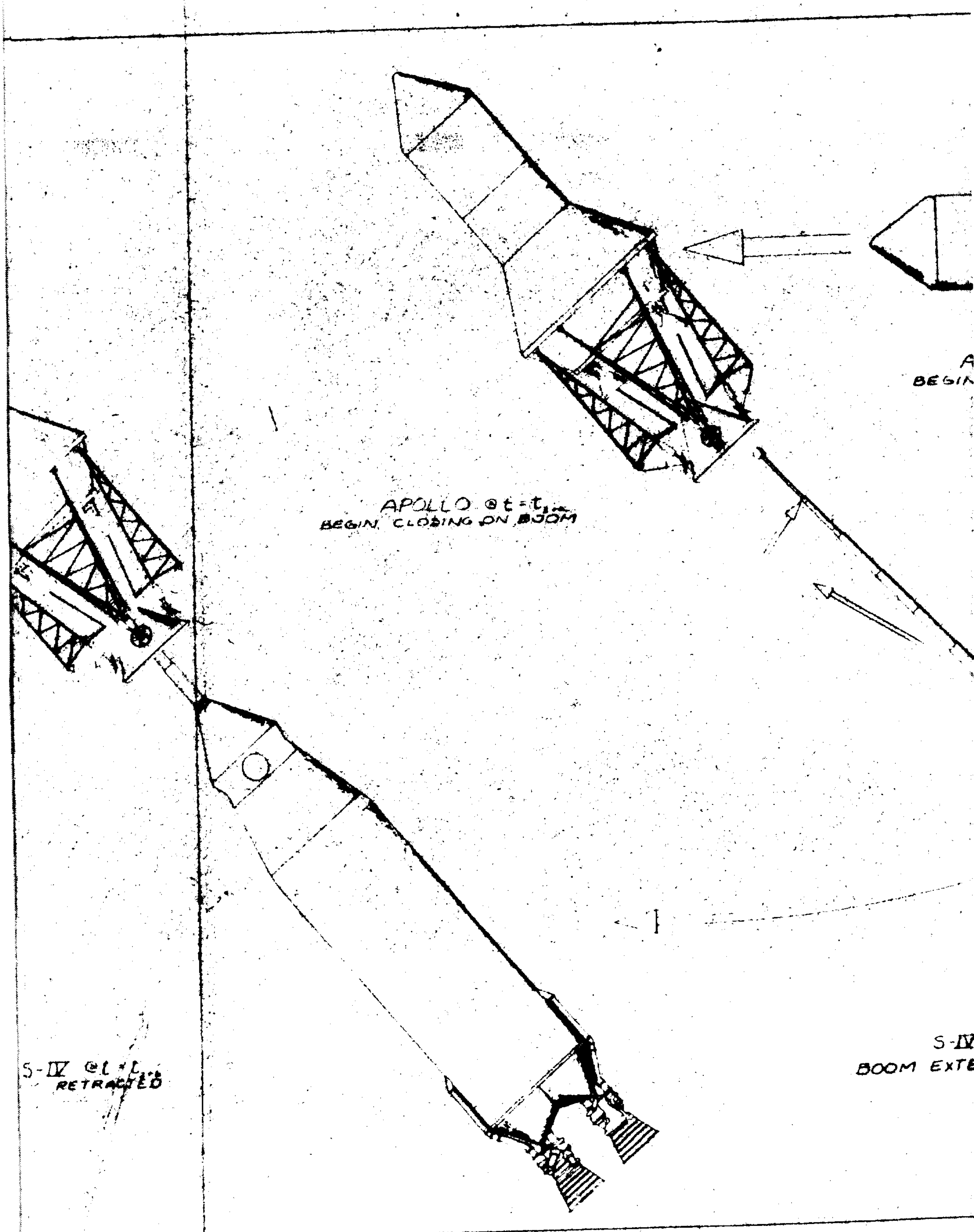


S-IV

APOLLO [illegible]
LUNAR LANDING MODULE ALIGNMENT GEAR DEPLOYED.
LLM ENGINES FIRE TO TRANSLUNAR INJECTION VELOCITY
CUT-OFF AS A SECOND STAGE TRANSLUNAR INJECTION BOOSTER.
VERNIER ENGINES ARE FIRED FOR TRANSLUNAR MID-COURSE
CORRECTION. LLM ESTABLISHES CIRCUMLUNAR ORBIT AND
PERFORMS LUNAR LANDING.



FOLDOUT FRAME 2



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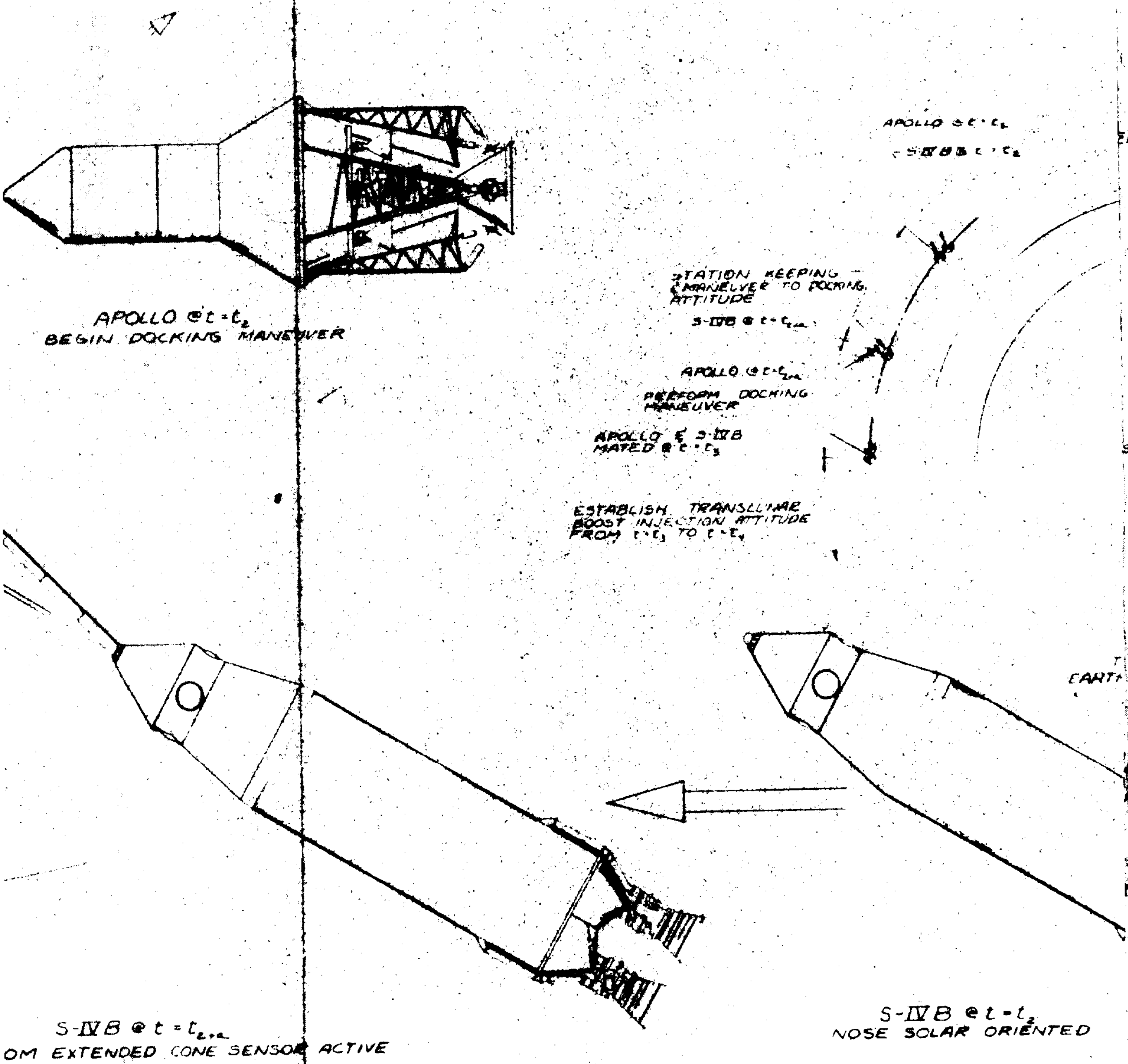

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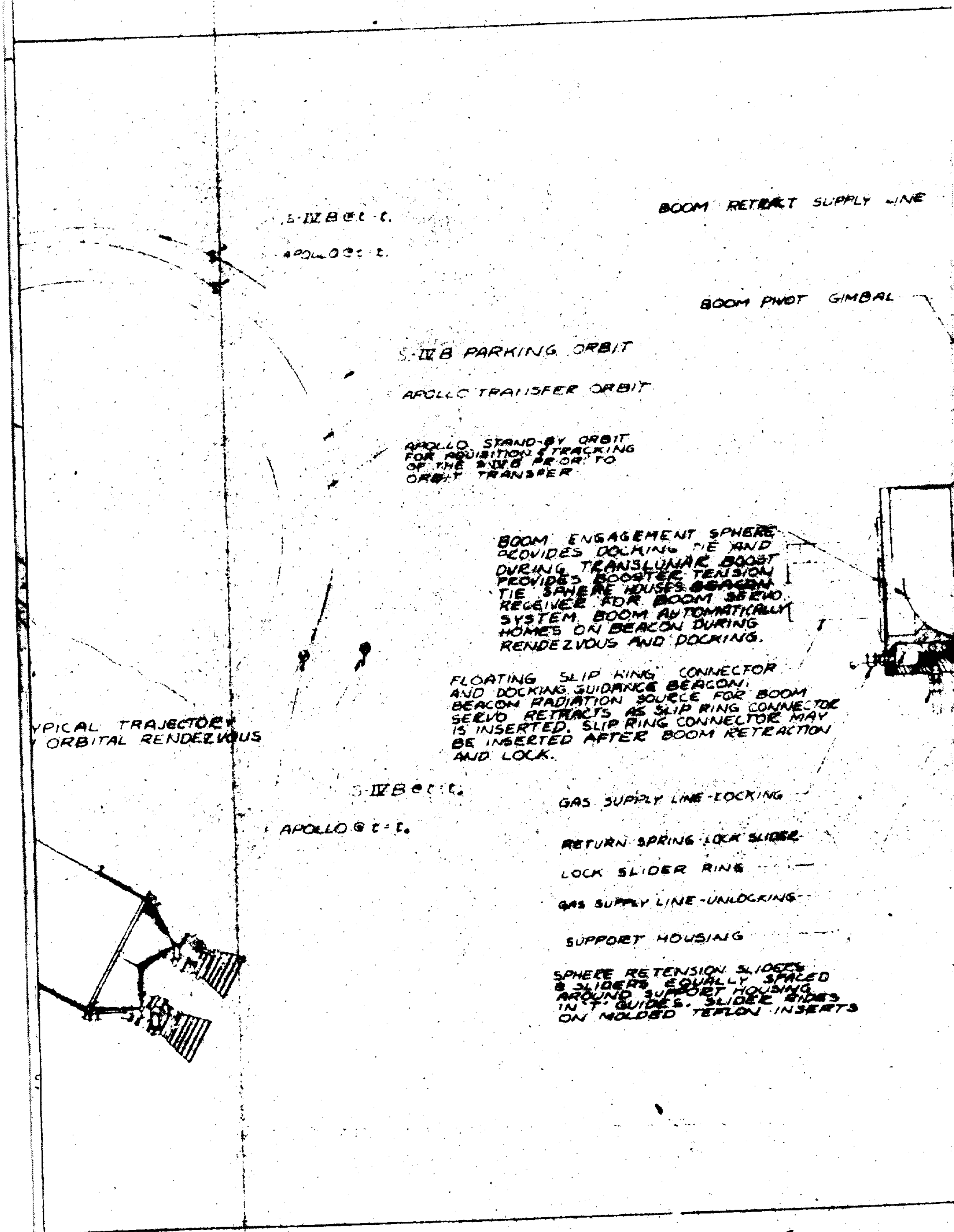

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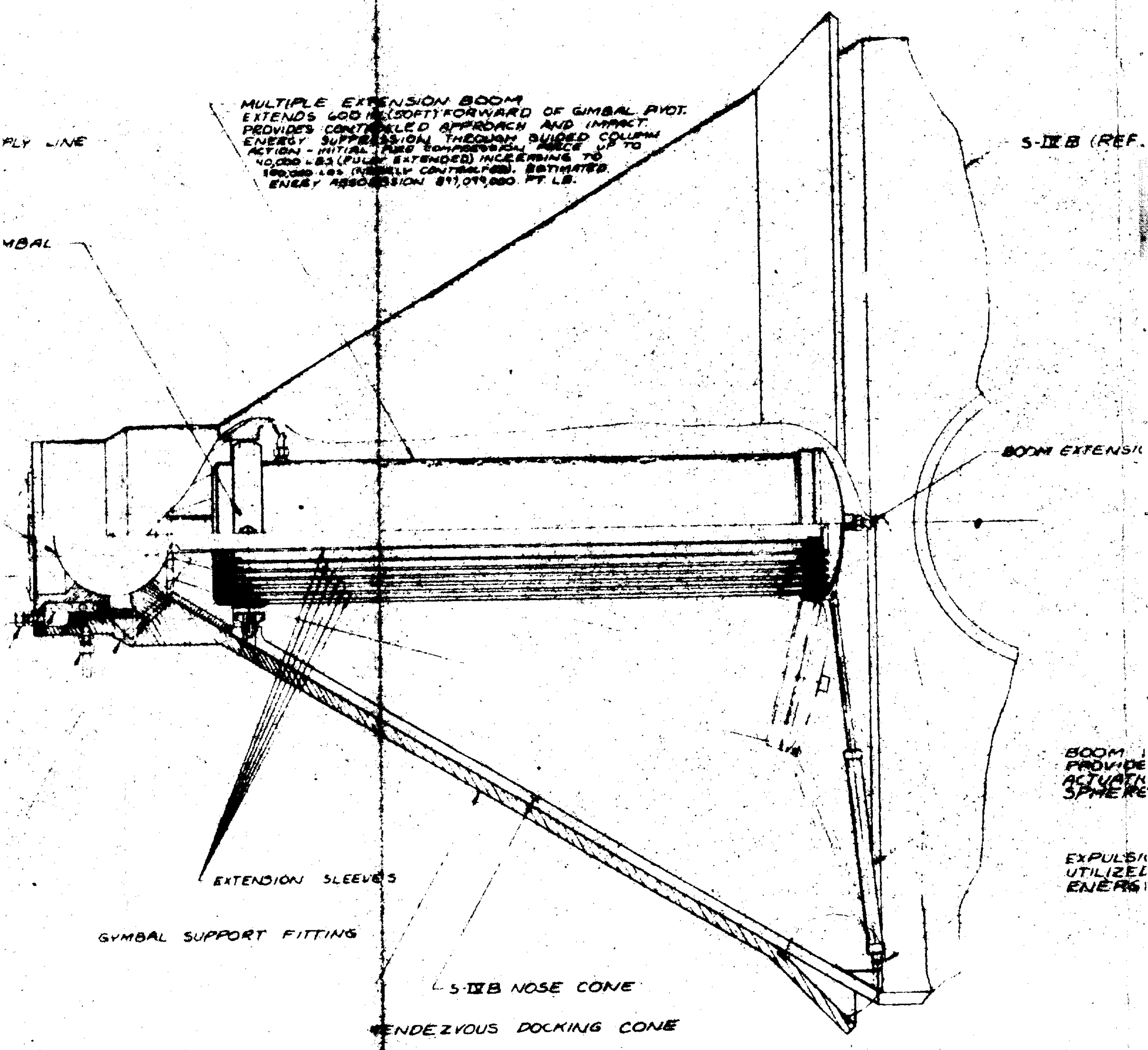
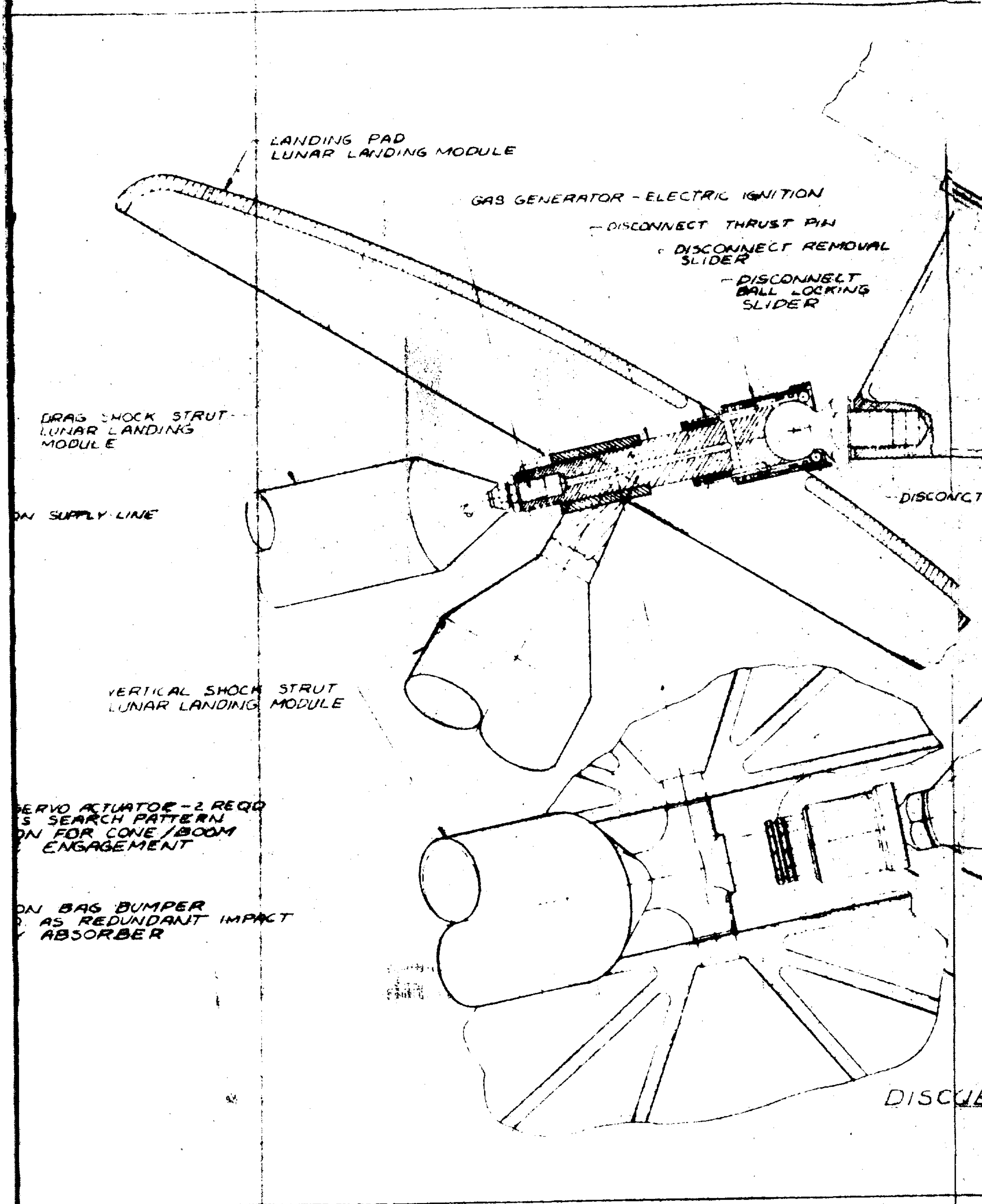


RENDEZVOUS MECHANISM DETAILS

FOR LLM LANDING GEAR AND DISCONNECTS SEE DISCONNECT DETAILS
SCALE - ONE TENTH SIZE

FOLDOUT FRAME 6



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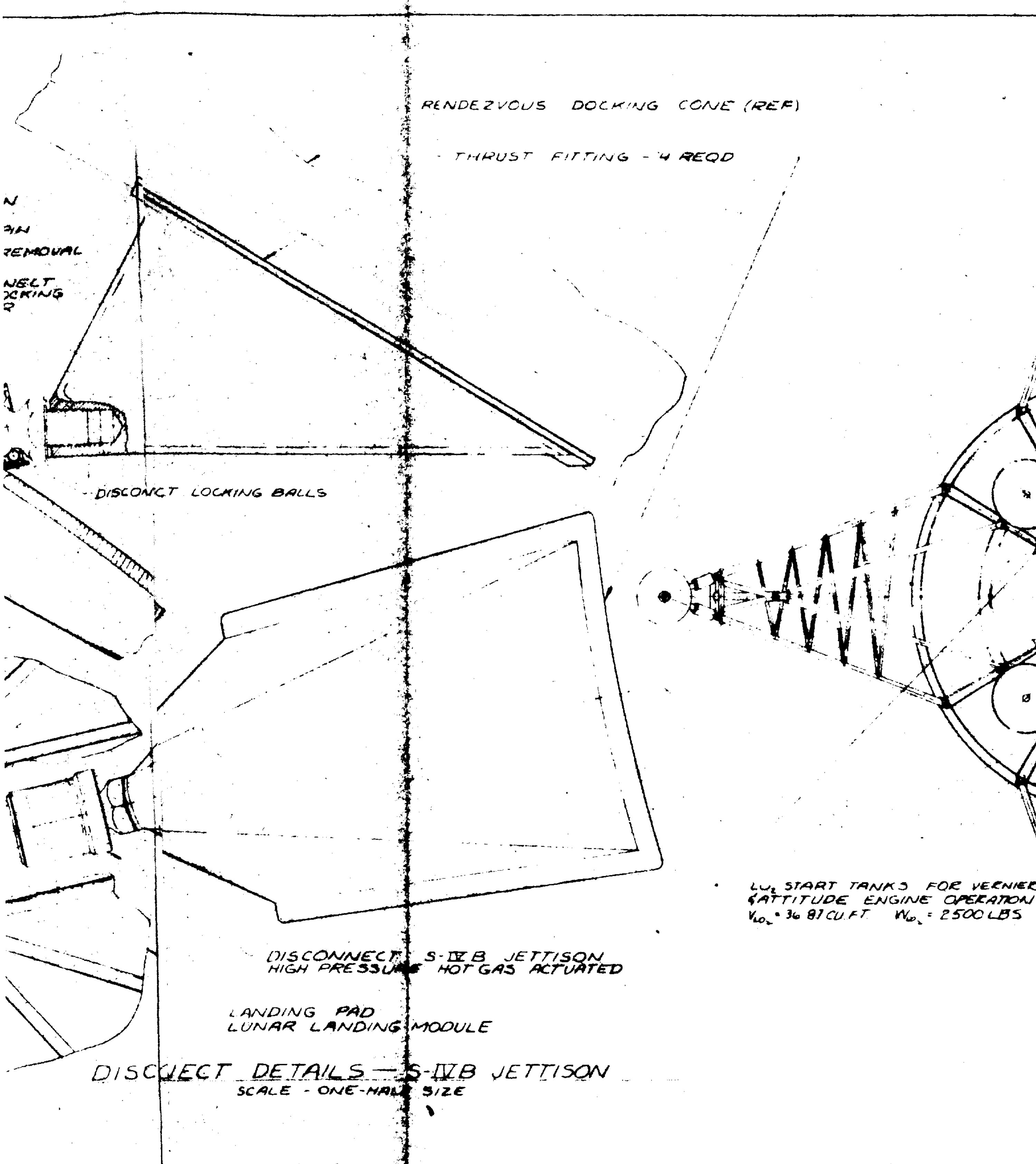
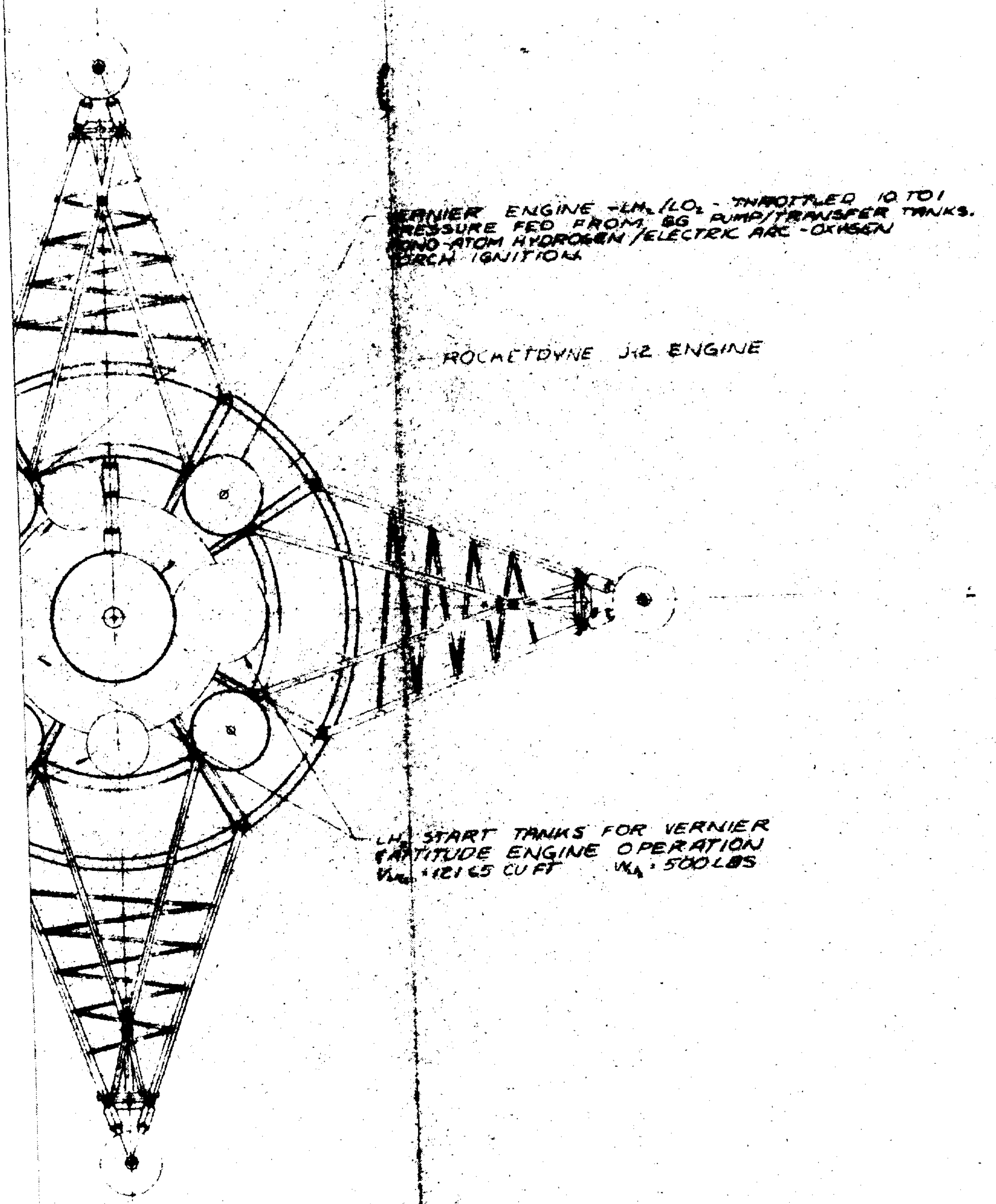
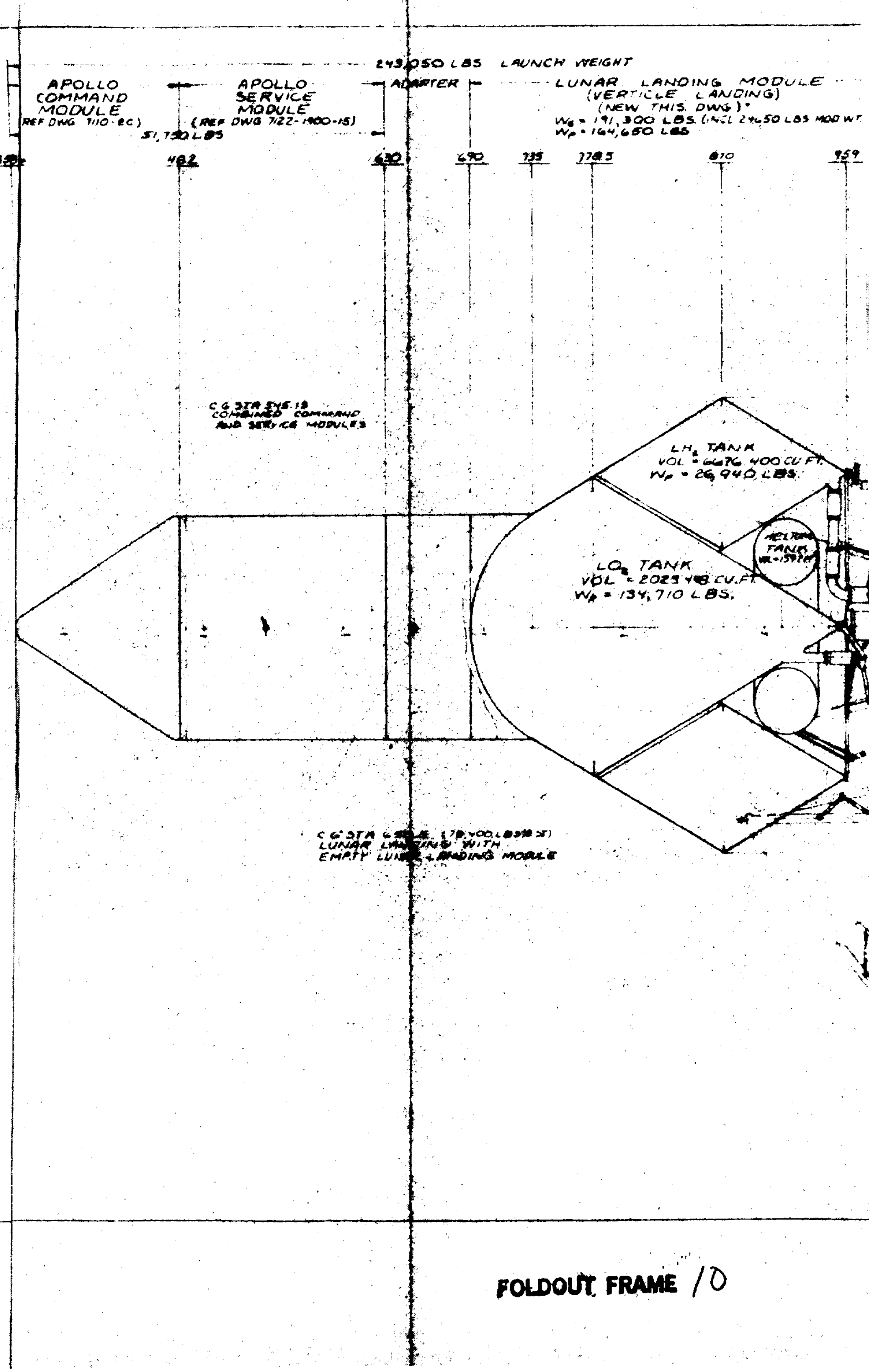


FOLDOUT FRAME 8



FOLDOUT FRAME 9

LED 10 TO 1
NSFER TANKS.
OXYGEN



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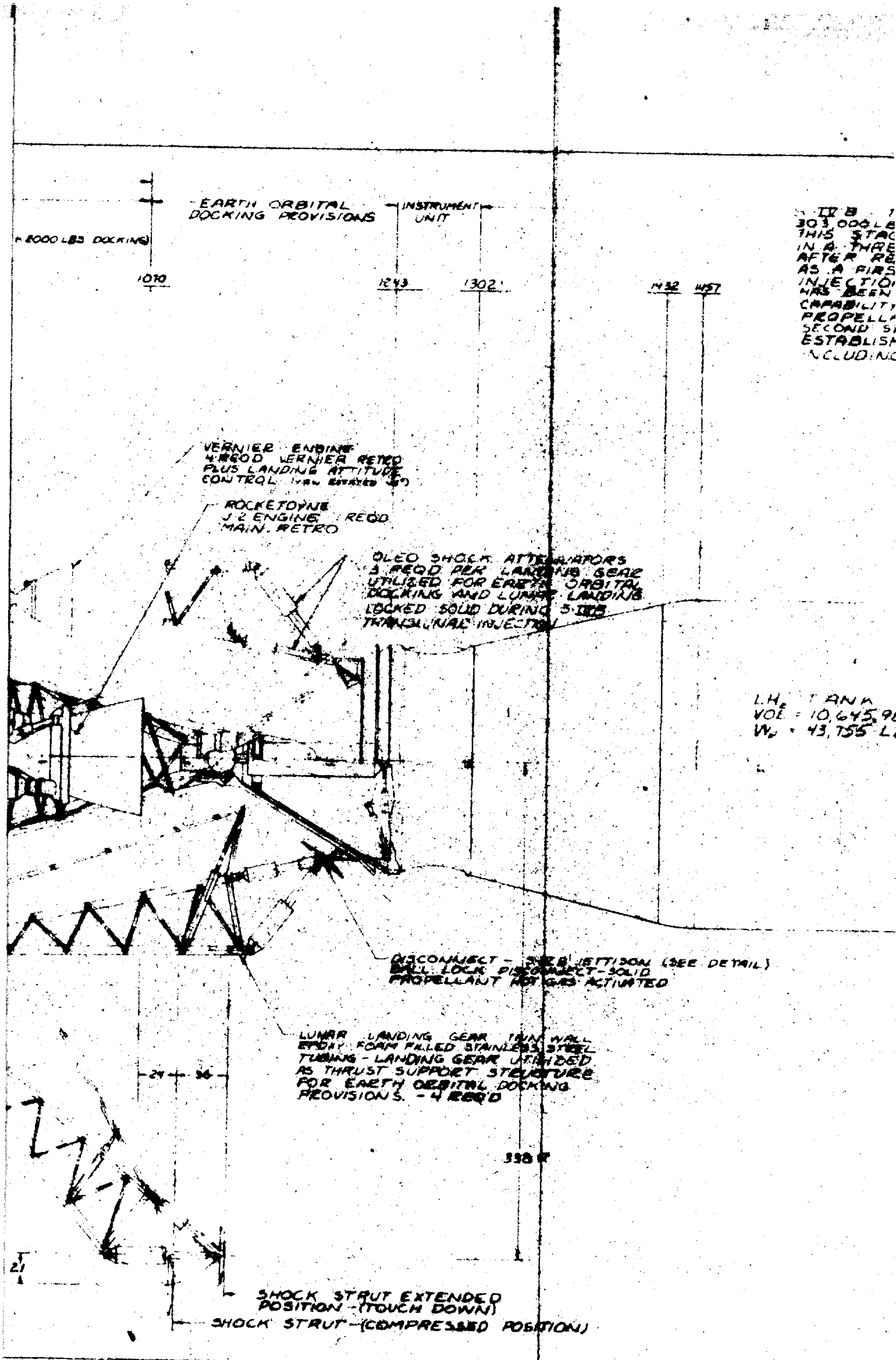


FOLDOUT FRAME 11

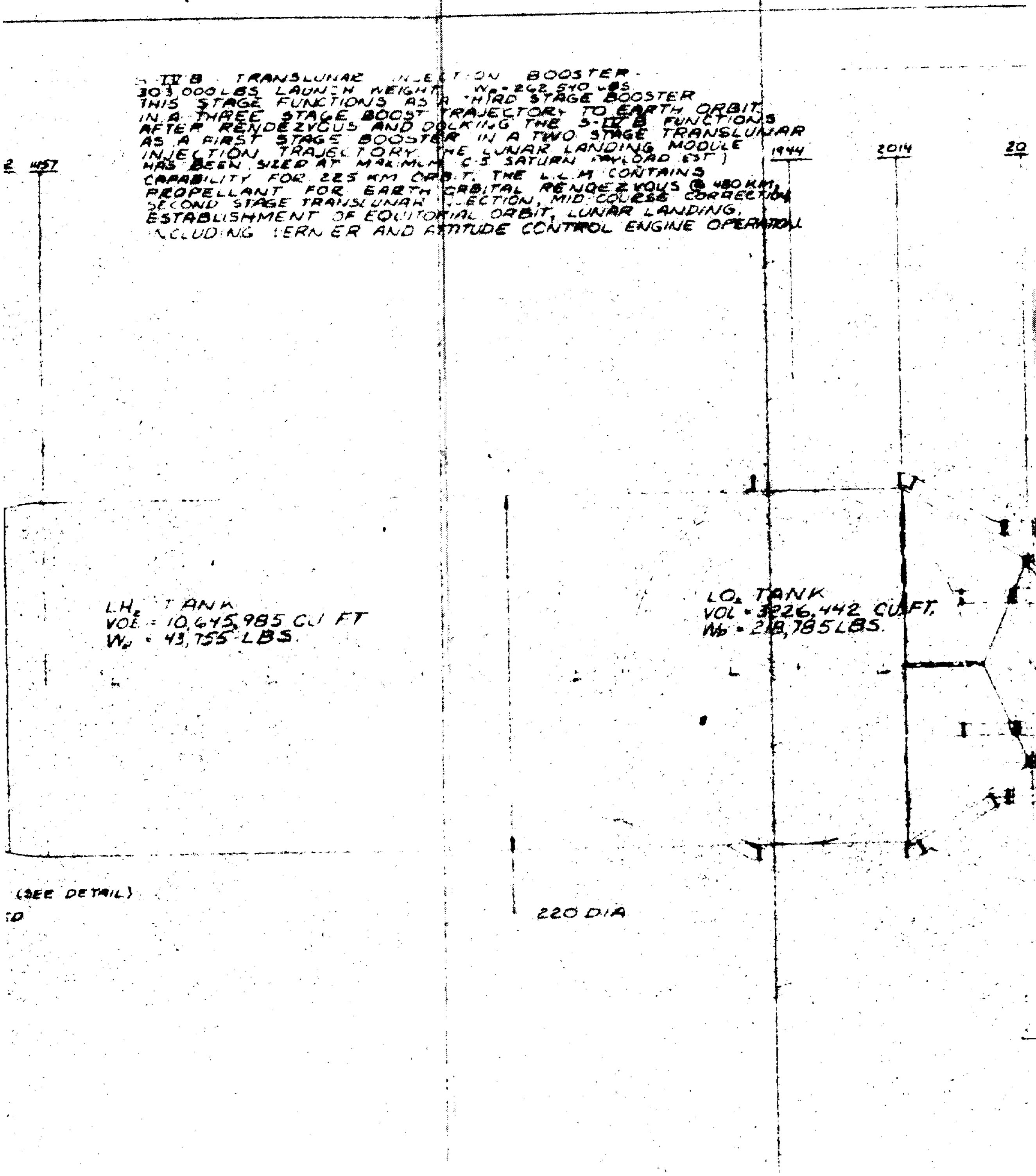
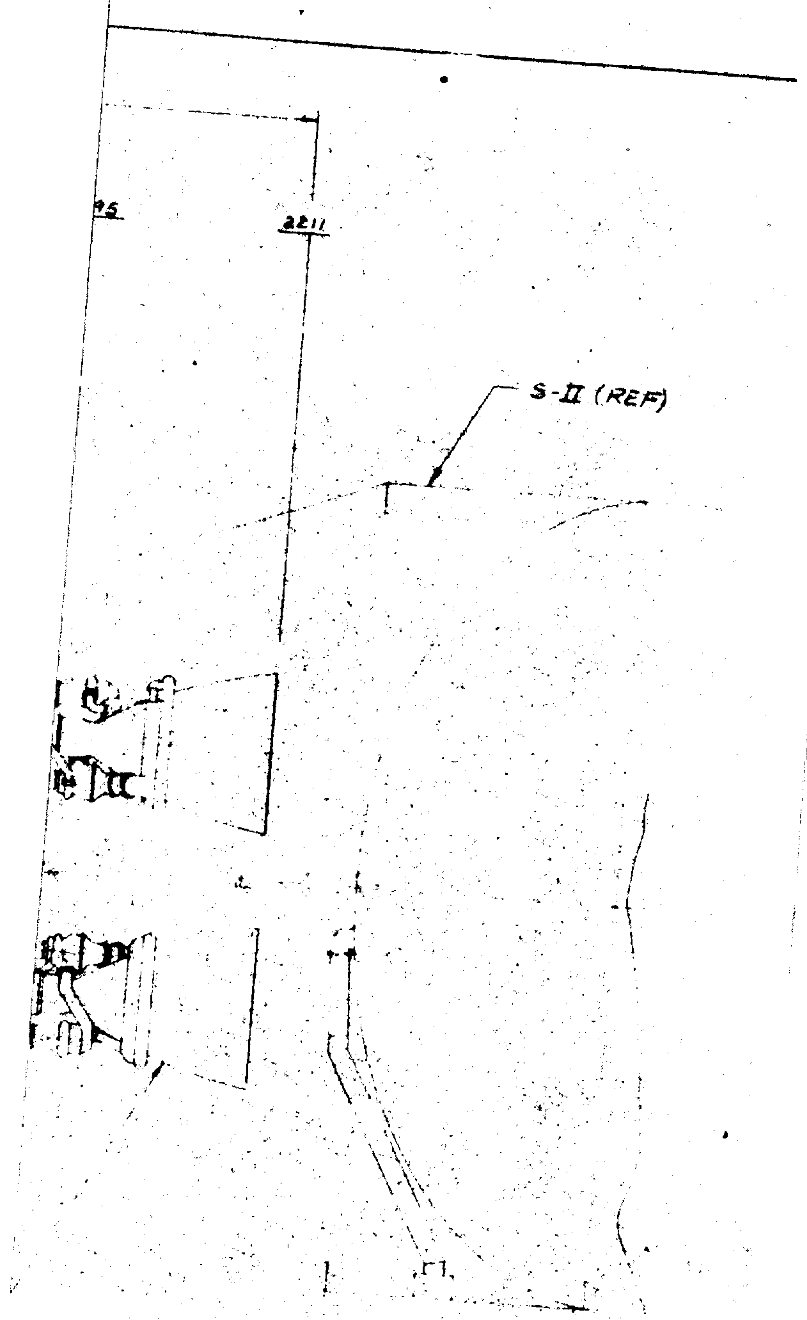


Figure 5-1.
Rendez

FOLDOUT FRAME 12



Docking S-14B and Apollo Earth Orbital
...vous Method and Mechanical Details

5-11, 5-12

FOLDOUT FRAME 13

APPENDIX

GEOPHYSICAL DATA, TRACKING STATIONS, APOLLO RENDEZVOUS

In the body of the report, capabilities of the proposed additional Mercury tracking stations were evaluated on the basis of line-of-sight conditions. The validity of such an evaluation is dependent on the geophysical properties of the terrain at the various locations. An attempt will be made to show that the assumption of line-of-sight conditions for tracking is not seriously degraded by the judicious selection of tracking sites.

Orbiting of S-IVB and Apollo is contained within 32nd north and south parallels. A predominant number of tracking stations on the proposed network will be located near the north and south 30 degree parallels. Consequently, about half their coverage will be in areas of little orbital activity. Coverage of each station and possible pattern nulls due to geographical consideration will be discussed in this Appendix.

Cape Canaveral is located on a low isthmus whose elevation is seldom higher than the antenna height. Consequently, line-of-sight conditions will exist for all azimuth angles, except possibly to the west where trees may cause some low-horizon problems. However, the proximity of the Eglin tracking station will negate any problems. Mountainous areas in eastern United States are all above 32 degrees latitude and pose no problems.

Bermuda, located at 32 degrees, 12 minutes north, is a relatively low island. Its station is located on the south end of the island.[1] Therefore, unobstructed line-of-sight conditions will exist throughout the important southern azimuthal angles.

Station No. 4 is located on the south side of Grand Canary Island.[1] Since the Canary Islands are volcanic in nature, the center of the island should be at higher elevations than the shore. This could cause a line-of sight blackout to the north. However, only two orbits pass north of the station and both will be less than 4 degrees north of the station. At vehicle altitudes of 90 miles or more, there will be no blackout and tracking will be continuous.

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1. Weaver, K.F. and R.F. Sessions. "Tracking America's Man in Orbit," National Geographic, Vol. 121, No. 2 (Feb. 1962).

Station No. 5 is at Kano, Nigeria. Kano is located on a plain at an elevation of slightly over 1000 feet. There are higher elevations to the south and southwest.[1] Again, these elevations do not cause problems at orbital altitudes of 90 miles and more; minimum vehicle elevation angle is still larger than maximum obstruction elevation angle.

Zanzibar Island is station No. 6. The station is located inland near the village of Tunguu. Higher plateaus on the African mainland to the west will cause line-of-sight to be raised to almost 1 degree. This may cause Zanzibar to miss tracking time on the ninth orbit of a 90-nautical-mile-altitude pass but this particular orbit would have been tracked by the South American ship station and would next be tracked by the Gulf of Aden ship, which is less than 180 degrees of longitude away.

Station No. 8 is located at Muchea, 35 miles from Perth on the West Australian coast. It has unrestricted view in the western hemisphere. To the east is slightly higher land but of insufficient height to cause problems. In this area there is also an overlap of coverage with Station No. 9 at Woomera. Station No. 9 is on a flat, desolate, desert area whose nearest geographical obstruction is 5200-foot Mount Woodroffe, approximately 350 miles to the north-northwest. This is a point obstruction of less than 1 degree in elevation and will cause no blackouts of orbits passing to the north due to relatively high vehicle-elevation angles. Slightly more than 250 miles directly east of Woomera is the Flinders Range with maximum elevation of 5000 feet, which will raise line-of-sight angle to almost 0.3 degrees. This will cut Woomera's tracking orbits 14 and 15 short by seven seconds maximum, less than 3 percent total tracking time. Both orbits will also pass through Noumea coverage.

Canton Island is a coral atoll in the central Pacific. Telemetering antennas are mounted on a tower[1] so that line-of-sight conditions are met.

Station No. 12 is at Kokee on the island of Kauai in Hawaii. Its location is on a 3000-foot plateau[3] on the northwest portion of the island. Twelve miles to the southeast is 5080-foot Mt. Waialeale; elevation angle is slightly less than 2 degrees. This will decrease tracking range on a 90-nautical-mile orbit by 175 nautical miles (from 800 down to 625 nautical miles) and on the 300-nautical-mile orbit tracking range will be decreased by 225 miles (from 1465 to 1240 nautical miles). The only orbit whose tracking may be affected will be orbit seven, which passes northwest to

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2. Goode, J. P. Goode's School Atlas. New York: Rand, McNally and Company, 1950.
3. Range Manual. Pacific Missile Range Report, PMR-MP-60-9.

southeast just to southwest of the Hawaii chain. On a 90-nautical-mile orbit, track time may decrease 37 seconds maximum to about 5.1 minutes minimum of track and approximately 47 seconds maximum to 9.8 minutes minimum of track on a 300-nautical-mile orbit.

Station No. 13 is located at Point Arguello atop 2159-foot Mount Tranquillon, the highest point in the area. As this station is at 34 degrees, 30 minutes north latitude, it need only cover its southwest quadrant and partially cover its southeast quadrant because no orbits should pass over or north of this station. Its view to the east is limited by higher elevations of the coast range. However, Point Arguello's coverage pattern overlaps that of Station No. 14, Guaymas, so no coverage null will exist. Guaymas has an unrestricted view to the west and south (coastal mountains in Baja California across the Gulf of California are at least 125 miles distant at an elevation angle 0.5 degrees). To the east and northeast of Guaymas is the high Sierra Madre plateau, which will shade the station in this direction. However, this deficiency is filled by Station No. 15 at White Sands and Station No. 16 near Corpus Christi, whose coverage overlap those of each other and Guaymas to prevent any lapse in coverage. Corpus Christi's coverage overlaps that of Station No. 17 at Eglin, which, in turn, overlaps coverage at Cape Canaveral. Consequently, these six North American stations present complete coverage from the Pacific into the Atlantic.

The line of sight of all ship stations, such as the Rose Knot in mid-Atlantic and Coastal Sentry in the Indian Ocean, is restricted only by the horizon. The same is basically true of the three new ships proposed for the west South American coast, Persian Gulf, and Gulf of Aden. The Gulf of Aden's station could be located on the British island of Sokotra, which has a high mountain on its eastern end. If the station were located on the western tip some shading would result on the eighth orbit, but this orbit also passes through the center of the Zanzibar zone just previous to entering the southeastern area of the Aden zone.

The proposed new station at Noumea could present no particular geophysical problems because New Caledonia is a relatively low island. The station located at Durban, South Africa, would have shadowing to the west and up to the west-southwest. In this area, there is a 10,000-foot mountain range topped by 10,761-foot Mont aux Sources. This mountain range would decrease tracking radius on a 90-nautical-mile orbit by 140 miles and on a 300-nautical-mile orbit by 165 miles in this direction. On orbits three and four, tracking time will be decreased 30 seconds (from about 5.7 to 5.2 minutes) on a 90-nautical-mile orbit and 35 seconds (from about 10.3 to 9.7 minutes) on a 300-nautical-mile orbit. Orbit seven, which passes from southwest to northeast northwest of Durban, will also have its tracking time shortened; but this orbit will be tracked by Zanzibar station.

ORBITAL RENDEZVOUS ELECTRONICS SYSTEM

The study of electronic techniques to effect orbital rendezvous is based on certain assumptions. Early information indicated that S-IVB (or similar booster) would be boosted into a 300-nautical-mile parking orbit followed by the launching of the Apollo vehicle into a 90-or 100-nautical-mile parking orbit. At the appropriate time when the two orbits are nearly parallel, the Apollo will be boosted into the higher S-IVB orbit and ahead of the mating booster. Orbital velocity of the Apollo will then be decreased by small retro rockets to allow the S-IVB booster to approach the Apollo vehicle. At the appropriate time, Apollo velocity will be increased to allow mating of the two vehicles at zero, or near zero, velocity to prevent any structural damage to the two vehicles and to minimize structural problems in vehicle design.

Satisfactory guidance systems to effect parking orbits are available. The guidance system of the Apollo will also be able to place the vehicle into the appropriate 300-nautical-mile orbit, although it is not deemed necessary to place the Apollo into the same exact track as S-IVB. Therefore, it is assumed that the smaller retro rockets and/or vernier-type engines of the Apollo can provide limited attitude control of the vehicle. This attitude control system may be automatic with the capability of manual operation as an emergency back-up.

This study is based on the theory that the major portion of the system will be on the Apollo vehicle (which in this discussion includes the command module, service module, and lunar landing module) where it may be used for the secondary purpose of acting as both an altimeter and velocity sensor for lunar landing. The system, with certain modifications, may be usable during earth reentry if desired.

Sensor Selection

In the choice of a proper system to electronically effect rendezvous many factors must be considered. Weight, volume, and primary power requirements must be minimized and search range must be optimized. Readily available, reliable components must be used. As far as possible, the system should be able to function in more than one operation, possibly using components of other systems, to decrease total vehicle power and volume requirements.

Various sensors potentially usuable in space rendezvous were investigated by Heiss and associates [4] who concluded that only the active

4. Heiss, W. H. et al. Space Rendezvous Terminal Sensors. Procedings of the National Aerospace Electronics Convention (1961).

systems can provide the range and range rate data with the required degree of precision. Heiss' investigation assumed an uncertainty volume of 4 to 10 nautical miles in height, 2 to 5 nautical miles wide and 5 to 40 nautical miles deep with a maximum detection range of 400 nautical miles and closing velocities of 500 to 7000 ft/sec. Although the active noncooperative radar requires no equipment in the target and can discriminate between the target and other sources of radiation (except thermal noise), its greatest disadvantage is the need for voluminous equipment and large primary-power requirements to achieve the necessary detection range. A scanning, noncooperative radar system could require up to 2000 watts primary power and up to 150 pounds of equipment for maximum 1000 mile range; a non-scanning noncooperative radar could weigh up to 350 pounds and require 50 kw primary power for the same maximum range. Alternatively, an active cooperative radar could achieve similar ranges with only 90 pounds weight maximum and about 900 watts primary power for an interrogator employing angle search and track with the target transponder weighing about 8 pounds and using 5 watts primary power. Heiss also noted that with cooperative radar, system power, size, and weight are essentially independent of range. Thus, active cooperative radar has become the favorite system for orbital rendezvous and its development is being actively pursued by several companies[5, 6, 7].

Other rendezvous systems have also been investigated. The infrared systems present high resolution and optical gain at their short wave lengths but present state of the art will not permit accurate determination of range. Infrared systems could also suffer from flame attenuation, and location of infrared sensors would, therefore, be critical on the Apollo vehicle. Reflected sunlight systems are not satisfactory. Since the target would have to be illuminated by sunlight, the time and place of rendezvous would be restricted. Additional limiting factors include noise generated at the detector by background radiance; false targets presented by bright stars; image orthicon detectors which have low information rate per picture element; and MTI techniques that may be necessary to discriminate against star background. Although star background problems can be eliminated in

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5 Getler, M. "Radar Beacon Proposed for Rendezvous," Missiles and Rockets (12 March 1962), 31.

6 Bonelle, G. J. Radar Design for Manned Space Vehicles, presented at the Military Electronics Convention, Los Angeles (Feb. 1962).

7 Reuter, H. A. Radar System for Unmanned Cooperative Rendezvous in Space, presented at the IRE International Convention, New York (27 March 1962).

thermal radiation systems, their detection range is limited by detector noise. Thermal detectors also have a low information rate per detector; and, therefore, such a system would require many detectors. Optical systems, particularly those using Lasers, are a possibility for the future but presently are too new and have too low efficiency to be considered. Woodbury[8] reports obtaining a seven-mile range on a clear day with an experimental coherent light detection and ranging (COLIDAR) system whose power conversion efficiency is less than 1 percent, while Stitch[9] earlier reported a three-mile range against white stucco with a similar system whose beam width was only 0.3 milliradians wide. Although presently unacceptable, Lasers should be considered for the future; advances have already been reported in CW lasers[10] and in the powering of lasers with solar radiation alone[11].

Radar System Selection

Selection of a proper radar system to use during space rendezvous should be based on system simplicity and minimum volume, weight, and power requirements. The possibility of being able to use the rendezvous system, or at least parts of the system, to meet other requirements of the space vehicle should not be overlooked.

Of particular interest in the selection of a radar system is the choice of frequency and antennas for operation. Within limits, weight and volume requirements are dependent on frequency. Mueller[12] has shown that the total weight of a system using parabolic antennas increases with frequency for a given transmitted power. This is based on a decreasing efficiency of

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8 Woodbury, E. J., et al. "Design and Operation of an Experimental COLIDAR." Proceedings Western Electronics Show and Convention, 1961.

9 Stitch, M. L., et al. "Breadboard COLIDAR." Proceedings Military Electronics Convention, 1961.

10 Vogel, S. and L. Sulberger. "Crystal Laser Puts out Continuous Power." Electronics (12 Jan. 1962).

11 Maguire, J. "What's Ahead for Optical Masers?" Electronics (March 1962).

12 Mueller, G. E. "A Pragmatic Approach to Space Communications," Proceedings of IRE, Vol. 48, No. 4 (April 1960).

power conversion of available transmitters, increase in frequency plus antenna gain being proportional to the square of the antenna's diameter in wave lengths and its weight being proportional to the antenna's area. Lorens[13] shows that with directional antennas and the maximization of effective radiated power, the antenna weight is two-thirds the transmitter weight. This weight problem is borne out in a report[14] that notes that a 36-inch, K_u-band parabolic antenna presents two-thirds of the total weight of the rendezvous (non-docking) system and approximately 37 percent total system weight, including docking transponders and their antennas and rendezvous radar equipment. Therefore, high-gain parabolic antennas should not be used because of their large weight and size. The need to rotate parabolas will add other problems. It is recognized there will be two 54-inch parabolic dishes on Apollo for deep-space communications in the 2100 to 2300 mc region. However, additional complexity required of antennas for rendezvous, including the probable requirement to retract the units before boost into lunar orbit, would decrease their reliability for their primary purpose.

Flame attenuation problems must be considered in the choice of frequencies. Since retro-rockets and probably small vernier engines will be used to control the Apollo vehicle for space rendezvous, location of antennas is of primary importance. To assist in overcoming this problem, the operating frequency must offer minimum flame attenuation. Flame attenuation problems reported by Poehler[15] noted free electron concentration at the exit plane of a ballistic missile exhaust on the order of 10^{10} electrons per cm^3. When this information is used in the plasma frequency formula[16], it is found that the plasma frequency is 880 mc. These calculations are verified by Dirsa[17] whose curves also show slight attenuation for this electron density up to about 5000 mc/s, as well as an increase in propagation phase shift above the critical plasma frequency.

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13 Lorens, C. S., "Antenna Size for a Space Vehicle." Proceedings of IRE, Vol. 49, No. 11, November 1961.

14 Getler, M., op. cit.

15 Poehler, H. A. "Exhaust Flame and Antenna Breakdown Problems in the Transmission of Telemetry Signals during Ballistic Missile Powered Flight." Proceedings, PGSET National Convention, 1960.

16 Biggs, A. W. "A Method of Measurement of Flame Attenuation at 200 mc." Proceedings of the IRE, Vol. 49, No. 12 (December 1961).

17 Dirsa, E. F. "The Telemetry and Communication Problem of Re-Entrant Space Vehicles." Proceedings of the IRE, Vol. 49, No. 4 (April 1960).

Previous discussion concerns free electron concentration in an engine's exhaust that also contains contaminants from fuel. While testing Hawk motors, Raytheon [18] found that flame attenuation increases about 1 db for every 1 percent of aluminum in the propellant at S-band frequencies. Raytheon also cooperated with Reaction Motors, Inc., in a test of their proposed liquid-fuel motor for the lunar landing vehicle at a frequency of 4300 mc/s per second. No signal attenuation was detected at a distance of 5 feet from the nozzle. Although solid propellant rocket motor exhaust plumes will blossom in a near-vacuum environment, the metal oxides contained in the exhaust products will be confined to an optical path and will possibly cause attenuation in the exhaust core only [14]. Flame attenuation of the exhausts of the engines chosen for the lunar landing vehicle, including engines and/or rockets used in rendezvous, should be further investigated.

When general antenna and frequency information have been gathered, the specific type of radar system must be chosen. The types of radar systems considered were pulse Doppler, pulse, and FM/CW systems. Pulse doppler systems are used mainly to obtain isolation between transmitter and receiver that can be achieved in other systems by use of a transponder; further, pulse doppler cannot affect velocity lock-on and tracking for low relative velocities, and usually the system is heavy, more complex, and costlier. The choice between pulse and FM/CW systems is arbitrary. There are proponents for both systems [19, 20]. Both systems also use standard, proven circuitry and components; and both use transponders. A comparison of the two systems is shown in Table A-1, [21].

The FM/CW system can be used for lunar landing [22] by adding twelve pounds (for antenna gimbals) and 125 watts primary power. The two systems weigh about the same. The FM/CW has greater range at a cost of a greater requirement in transponder power. The FM/CW system is slightly more accurate, particularly at minimum range. It may have a

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18 Correspondence with Raytheon Company

19 "Radar Systems for Orbital Rendezvous and Lunar Landing." Raytheon Company, Missile and Space Division, Report BR 1396, October 1961.

20 "Cooperative Satellite Rendezvous System." Westinghouse Electric Corp., Air Arm Division, Technical Memo AAN 380 11-4 (16 October 1961).

21 Refer to 19 and 20.

22 Ibid.

Table A-1. Comparison of Pulse and FM/CW Radar Systems

Parameter	Pulse	FM/CW
Maximum range	60 miles	200 miles
Minimun range	3 feet	5 feet
Radar weight	16.5 pound	22 pound
Transponder weight	13.3 pound	8.5 pound
Radar power input	25 watts	25 watts
Transponder power input	9 watts	20 watts
Angular accuracy	2.5 mils	1/4 degree
Range accuracy	±1 percent or ±20 feet	±1 percent or 2 feet
Velocity accuracy	±1 percent or ±1 feet/per second	1/2 feet/per second

capability for use during earth re-entry, if radomes can be provided for protection from plasma effects. This versatile system is recommended for radar sensing for space rendezvous.

Radar System Parameters

Interrogator System

The K_e band interrogation radar will be located in the Apollo vehicle, its exact placement depending upon the multiplicity of functions it must perform. For use as rendezvous radar alone, or for use in rendezvous and lunar landing, the system may be placed in the lunar landing vehicle. If the system is also called upon to perform during atmosphere reentry, it must then be located in the Apollo command module. This necessity would result in additional location problems, not the least of which is antenna location.

The FM/CW interrogator will weigh approximately 34 pounds and consume 150 watts of power while transmitting 6 watts. This klystron type of system would be capable of both rendezvous and lunar landing missions and could be used for earth reentry if minor improvements are made. The performance of this system should be in accordance with the following chart:

Table A-2. Interrogator Performance

Parameter	Rendezvous	Lunar Landing
Maximum range	200 nautical miles	20 nautical miles
Minimum range	5 feet	10 feet
Velocity accuracy	0.5 feet per second (0.1 second smoothing)	1 foot per second or 0.1 percent
Angular accuracy	0.015 degrees	
Angular rate accuracy	±.02 degrees per second (1 second smoothing)	
Range accuracy	±2 feet	±2 feet

A block diagram of the combined lunar-landing rendezvous interrogator is shown in Figure A-3. The basic carrier frequency is generated in the microwave oscillator and is fed to a power splitter where it is split for use with lunar-landing velocity measurements. Part of the power for the phase modulator is picked off and fed to a single sideband generator (SSBG). The ranging modulator has outputs of 2 mc/s, 20 kc/s and 800 cps corresponding to maximum ranges of about 0.8 miles to 200 miles. The 2 mc/s signal is multiplied by 12 and fed to the SSBG where it is impressed upon the basic frequency and fed through position 1 of switch 1 and position 2 at switch 2 to the rendezvous antenna. The received signal is obtained through the same side-stepped antenna at a frequency of 24 mc/s ± the doppler frequency higher than the basic transmitted frequency. This frequency shift makes possible separation of transmitted from received signal. The received signal is fed through the mixer to a 24 mc/s IF amplifier and thence to one channel of the frequency tracker. The frequency tracker uses a variable frequency oscillator and fixed filters to track the carrier sidebands. the filters are heterodyned with the 24 mc/s reference frequency to produce a doppler frequency which is measured by the counting of cycles to obtain velocity. The carrier and sidebands are also passed to the ranging and phase comparators where the ranging frequency sidebands are extracted and phase compared with the original modulation frequencies to obtain range data. Angle data is obtained directly from the gimballed antenna system.

For lunar landing, switch 1 is in position 2 and switch 2 is in position 1, settings that cause the SSB generator (used only in rendezvous operations) to be bypassed. The interrogator then operates as a standard radar system. The outputs of the power splitter are fed to three feed-horns on the transmit antenna, one output passing through a phase modulator where it is modulated with range modulation signals. The three feed-horns are used to provide three-coordinate velocity information; one feed-horn is also in the range data system. The receiving horn is similar. The range and velocity inputs are fed through the IF amplifiers into various channels of the frequency tracker where they are used as previously described. However, there are three "velocity landing" digital outputs which are fed to an on-board computer (not part of the interrogator system) and thence to applicable control functions. One freqaency tracker channel is also fed directly through the frequency tracker to the range discriminator where range is measured as previously described.

Transponder System

The radar transponder will be located in the mating booster vehicle (either S-IV or similar booster). It is an 8-pound unit which consumes 20 watts of power. A block diagram of the transponder is shown in Figure A-1.

The incoming frequency modulated microwave signal is received at the antenna and mixed at the first detector with power from the stabilized local oscillator. The 2-mc/s sideband is extracted in a discriminator and multiplied to 24 mc/s. This frequency is then mixed with the complex 30-mc/s signal (which includes a doppler shift carrier and sidebands) to produce a 54-mc/s signal. This signal is then mixed with the stabilized local oscillator to produce a signal higher by 24-mc/s than the received signal (except for the doppler shift) which is then transmitted on the same side-stepped antenna. Gains on the order of 130 db through the transponder can be achieved.

The use of frequency offset between transmitter and transponder eliminates possible skin return problems and prevents reception of extraneous signals due to background noise or ionized layers of gas. The transponder also acts as a point source at the target and, as a result, eliminates near-range, angle, and range-track problems associated with glint noise. This frequency offset method is being used by most of the world's tracking stations.

Antennas

The proposed antenna system for use with the interrogator is shown in Figure A-2. It consists of three antennas mounted in a cylinder having a maximum diameter of 21 inches and a depth of 12 inches, all gimbals included. The transmit antenna is an elliptical metal plate lens with a major axis of 16 inches and a minor axis of 9.2 inches. The receiving antenna is also an elliptical metal plate lens with a major axis of 18 inches and a minor axis of

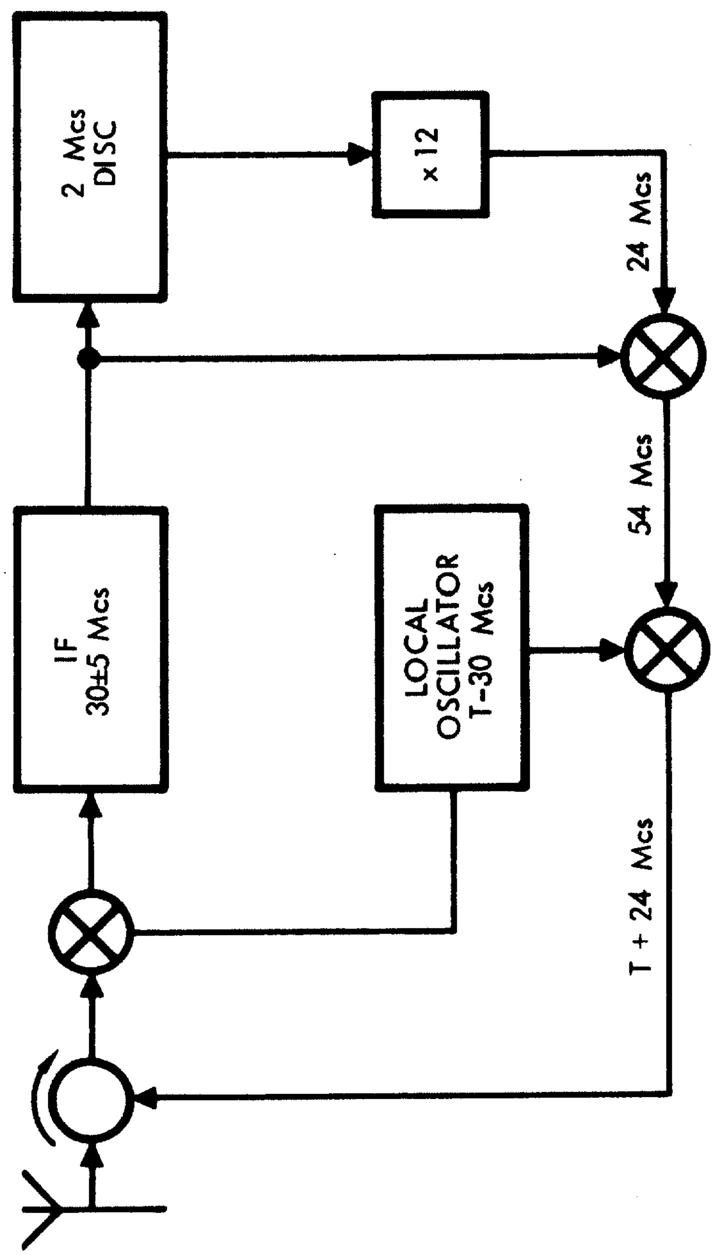

Figure A-1. Transponder

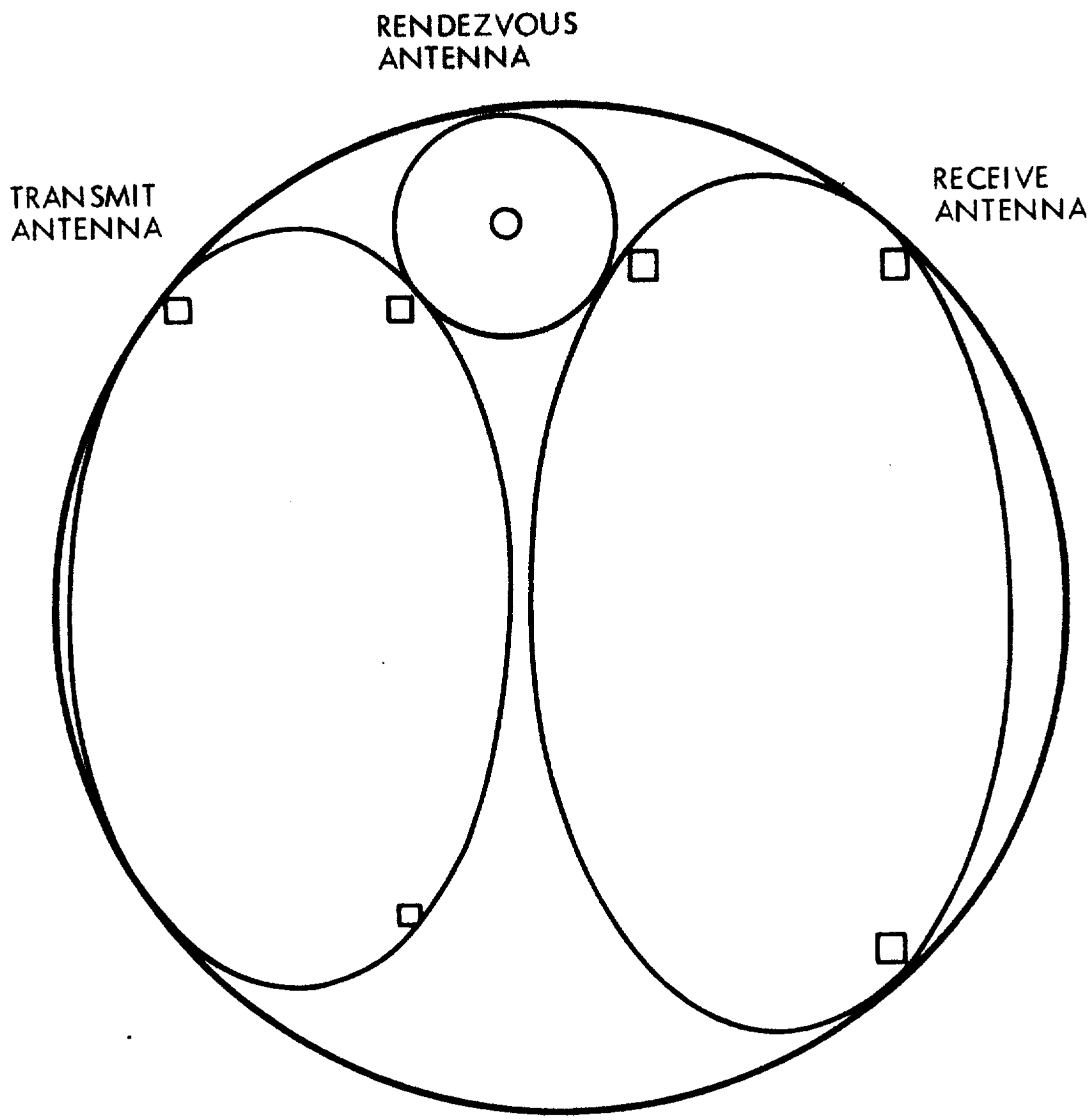

Figure A-2. Antenna System, Combination Radar

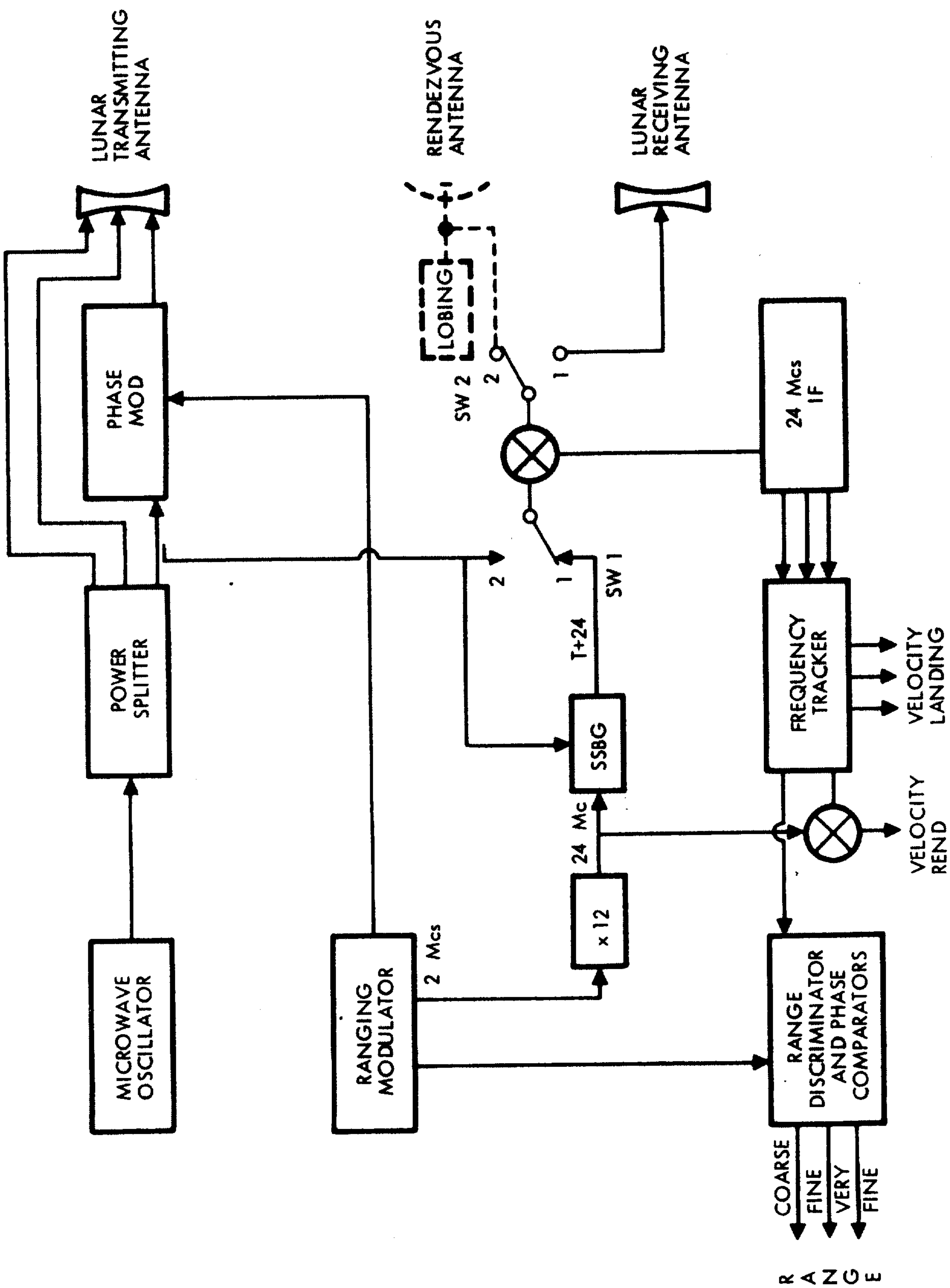

Figure A-3. Interrogator Radar

10.3 inches. The near field antenna is a 4-1/2 inch parabola used mainly for orbital rendezvous operations.

Each transmit-and-receive antenna has three feed horns as shown in the figure. Two of the beams are offset from the local vertical and measure two components of velocity for the lunar landing velocity sensor. The other beam points along the local vertical and measures another component of velocity as well as of range.

Depending upon its location, the antenna for the transponder may be omni-directional. Its most logical location seems to be on the "nose" of the booster because this will be the area that mates with the Apollo vehicle. In this location the antenna could be of cavity-backed slot design to give a hemispherical coverage. Such a pattern would permit an additional 2 or 3 db of gain to be picked up through the transponder loop.

Lunar Landing Capability

The previously described system has a lunar landing capability. No transponder, therefore, will be used and the interrogator radar will be employed in standard radar mode, the single sideband generator being bypassed by operation of switch 2(Figure 1). The interrogator will then "skin track" the moons surface to provide altitude and velocity data as noted previously.

For use in lunar landing, as well as in orbital rendezvous, the interrogator can be placed in the lunar landing vehicle. If located on the "bottom" (nozzle area) of the LLV, the antennas may need protection during docking operation. Protection may be provided, depending upon the docking operation required, by placing the antennas in a recessed area. The electronic components of the interrogator should be located as near as possible to the antenna in order to minimize transmission line loss.

Earth Reentry Capability

The FM/CW interrogator could be improved to include an earth reentry capability if such is required. Specific improvements would require an increase in transmitter power, greater sensitivity, and improved receiver noise figure.

Location of the interrogator to enable it to perform three functions will cause installation problems. From previous information[23] it is assumed that only the Apollo command module will return to earth. Therefore, in order to perform combined rendezvous, lunar landing, and earth reentry functions, the interrogator radar would have to be located in the crowded command module. Antenna problems would multiply. A single antenna

23 Rosen, M.N. and F.C. Schwenk. "A Rocket for Manned Lunar Exploration." IRE Transactions, Space Electronics and Telemetry, Vol. SET-5, No. 4 (December 1959).

system would be insufficient during rendezvous and lunar landing because of the "shading" brought about by the service module and the lunar landing vehicle; at least two antennas, diametrically opposed, would be required. The antennas would have to be flush-mounted during boost and capable of extension during operation; extension, depending upon the operation (lunar landing or preearth reentry), may require a "look" capability in either forward or aft hemisphere. During reentry the antennas would also have to be protected from deleterious plasma effects. Some type of switchable delay would have to be built into the transponder; if the antenna is located on the command module, a built-in range factor will be needed during lunar landing and rendezvous operations because of distance from the antenna to LLV.

A good inertial guidance system capable of adequately controlling the space vehicle during earth reentry would negate the need for the interrogator radar during this phase of the mission.

MATING GUIDANCE AND CONTROL SYSTEMS AFTER RENDEZVOUS

Introduction

The tracking of space vehicles in 100 and 300 nautical mile orbits has been proven feasible, although additional ground tracking facilities will be necessary for minimum track capability every half-orbit. Electronic systems are available to effect orbital rendezvous and docking of these two vehicles at an approximate 300-mile orbit. Mechanical systems for actual docking and joining of the two vehicles do not appear to be a major problem. This short report will delve into ways and means of mating the two control systems into a single control system before the Apollo vehicle is boosted into lunar orbit. Since the boost vehicle will be uncoupled from Apollo after its energy is spent, no permanent mechanical couplings can be made during this mating; control switching will have to be accomplished by electronic or simple electro-mechanical means.

Discussion

Both the Apollo vehicle and the booster will reach a rendezvous point under separate guidance systems. The problem becomes one of making two guidance and control systems act as one. This problem cannot be considered purely on its own merits; mechanical mating considerations cloud the issue. Ideas on the mechanical mating of the two vehicles are as varied as the designers from whom the ideas come. Sky hooks, basket-ball nets, drogue systems, mechanical arms, and other ideas have been presented. Final choice of a mechanical mating system actually depends on the accuracy of the docking system.

The mating of the electrical portions of the control system can best be accomplished through radio frequency links. This will allow the mechanical designer the greatest latitude of freedom. The r-f link should be as simple and reliable as possible, use a minimum number of components and antennas, and, wherever possible, use available components from other systems. A minimum amount of information should flow between the two vehicles.

The exhaust nozzle of the booster engine will be gimbaled with two degrees of freedom. The position of this nozzle should be known by the guidance control system. Consequently, two sensors will be needed in this application, and two channels of information must be transmitted from the booster to the Apollo guidance system. A servo system will be required to correctly position the nozzle. Information to control this servo must come to the booster from the controlling guidance system. The signal to fire the booster engine must emanate from the Apollo vehicle and be transmitted to the booster. Therefore, a minimum of four channels of information must flow between the Apollo and the booster. Each vehicle needs two channels for transmission and two channels for reception.

Implementation of the above four-channel system may present some problems. It is assumed that the booster has telemetry systems that will be operating. Nozzle gimbal position can be transmitted on one of these telemetry systems if the system has an antenna whose pattern covers the Apollo vehicle. The radio command system can be designed to receive the nozzle servo-position signal and the engine fire signal. If the radio command system for the booster must be usable for destruct purposes, either more channels will have to be added to present off-the-shelf units or a separate r-c receiver, operating at a different frequency and specially encoded, will have to be used. Antenna location for this particular link may be somewhat critical since it must be able to see the Apollo vehicle.

Apollo instrumentation will become more complex. A two-channel telemetry system, with antenna, will have to be installed; and its output will feed into the guidance system. A small, low-power, two-channel command transmitter, with antenna, capable of accepting signals from the guidance system, will also have to be installed. Antennas must provide proper propagation coverage of the booster.

Depending on the mechanical techniques used to mate the two vehicles in space, it may be possible to use the rendezvous radar system rather than radio command/telemetry systems. Both the radar transmitter (in Apollo) and the transponder transmitter (in the booster) could use special modulation techniques to transmit their particular data channels. The radar receiver and the transponder receiver could have special filters to separate the particular information and pass it into the action channels. This system, however, does not lend itself to ready use where the booster and Apollo mate into one large unit; such a mating technique would cover the antennas on both vehicles and block radiation; or, if the antennas should line up in line, the receivers would be blocked from saturation by a large signal. Alternately, if the two vehicles are mated to fairly tight tolerances, it may be possible to use an open waveguide technique for signal transfer. Here the signals from the booster could be transmitted by waveguide or coaxial line to the nose of the vehicle and terminated in a covered waveguide (similar to a leaky wave antenna). The command vehicle could have a similar open waveguide to receive the data and transmit by coax or waveguide to the proper data processing equipment. The thin radome cover on the open waveguide would be utilized for aerodynamic stability and would present some loss to the signal to prevent receiver saturation. The open waveguide could be conical in shape to allow for mechanical mating tolerances and still transfer energy. Transmission power would only have to be a few milliwatts; actual transmitter and receivers, per se, may not be necessary.

Previous discussion is based on a minimum amount of information flowing between the two mated vehicles. Other information may be required; if so, it will increase the control switching problem. As more channels of information are required, the need for two separate telemetry systems operating at different frequencies becomes greater; one system would have a transmitter in the booster and a receiver in Apollo, while the other system would have a transmitter in Apollo and a receiver in the booster. Two additional antennas per vehicle would be required; they would be located so that their radiation patterns face each other (if the open waveguide system is used a dual system will still be required).

In light of the proceeding discussion, the following recommendations are made: The guidance system in the Apollo vehicle should be the controlling guidance system after mating. Consequently, power to critical guidance control command circuits in the booster should be cut off to effect change. Necessary data signalling between the two mated vehicles should be accomplished by a two-way radio frequency link. If relative, tight, in-line, mating tolerances between the two vehicles is possible, the open transmission line rf link may be used. If the two vehicles are mated loosely, a two-way telemetry system will have to be used.